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LIQUID HYDROGEN - HIGH ENERGY ROCKET FUEL

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Eugene A. Cernan

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by

Eugene A. Cernan

Lieutenant, United States Navy

Submitted in partial fulfillment of
the requirements for the degree of

MASTER OF SCIENCE
IN
AERONAUTICAL ENGINEERING

United States Naval Postgraduate School
Monterey, California

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from the

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ABSTRACT

Liquid hydrogen has been classed as a high energy fuel for rocket propulsion. A survey of the latest technical literature was made and the information compiled in a form which discusses the value of this fuel in propellant combinations. Thermodynamic performance, payload comparisons, advantages, disadvantages, problems, and relative merits of respective combinations and systems are presented. A discussion of rocket performance parameters is included as a basis for a more complete understanding of the information presented in the above-mentioned areas.

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TABLE OF SYMBOLS

a_p = albedo of planet

c_p = specific heat (BTU/lb-°R)

g = acceleration of gravity (ft/sec^2)

h = static enthalpy (BTU/lb)

k = specific heat ratio (c_p/c_v)

$\dot{m}$ = mass flow rate (lbs/sec)

p = pressure (psia)

q = heat (BTU/lb)

r = radius (ft)

t = temperature (°R)

w = weight flow rate (lb/sec)

z = planet position factor

A = area (sq in)

A* = nozzle throat area (sq in)

F = thrust (lbs)

H = pressure head (ft)

I_{sp} = specific impulse (lb_f - sec/lb_m)

J = work conversion factor (778 ft - lb/BTU)

M = mass (lbs)

MW = molecular weight

N = rotational speed (RPM)

P = pressure (psia)

Q = volume flow rate (cu ft/sec)

Q_H = heat flow (BTU/hr)

R = universal gas constant (ft - lb/mole - °R)

S = suction specific speed ($rpm - gpm^{\frac{1}{2}}/ft^{3/4}$)

T = temperature (°R)

U = peripheral velocity (ft/sec)

V = velocity (ft/sec)

W = work (BTU/lb)

σ = Stefan - Boltzmann constant (1.713×10^{-9} BTU/sq ft-hr-$°R^4$)

ϵ = hemispherical emissivity

η = efficiency

ρ = density (lb/cu ft)

Λ = propellant mass fraction

Subscripts:

a = ambient

c = chamber

e = exhaust

p = planet

s = sun

sv = net positive suction

u = tangential

B,BO= burnout

LO = lift-off

PRO = propellant

1. INTRODUCTION

The liquid propellant rocket is, and promises to remain for some time to come, the principal propulsion device for space exploration. The nuclear rocket, basically a liquid propellant rocket but with kinetic energy indirectly obtained from nuclear rather than by chemical reaction, is today several years from becoming an operational flight system. The scheme of electrical propulsion, although having exceptional merit for use in deep space, does not produce the required thrust to overcome strong gravitational fields; therefore, there is a dependence upon some method of insertion into this space environment. The large solid chemical rocket booster has relatively low development costs and is by comparison with its liquid counter-part, considered simple and reliable. However, when large velocity increments and a variety of missions are required, the superior performance and operational flexibility of the liquid propellant rocket system generally prevail.

In order to meet the requirements and future responsibilities of both military and large space boosters considerable research and development is being carried out in the handling the utilization of powerful liquid fuels and oxidizers. In the field of cryogenics great strides have been made and are being forecast using liquid hydrogen as the fuel and either liquid oxygen or liquid fluorine as the oxidizer.

The element hydrogen has long been recognized as a rocket fuel with outstanding thermodynamic performance characteristics. Its high heat of reaction with all oxidizers, combined with the low average molecular weight of the gaseous reaction products, produces specific impulses higher than chemical systems employing any other presently feasible fuels. The high diffusivity and chemical reactivity of hydrogen and its high cooling capacity simplify problems of injector and thrust chamber design. However, the element has some less favorable characteristics such as a very low boiling point and quite low density which make it necessary to study the payload capabilities of actual vehicles employing this fuel in order to evaluate its true worth. The

fact that hydrogen upper stages can significantly increase the payload placed in orbit by existing boosters has been recognized by the National Aeronautics and Space Administration in planning its space vehicle program. The liquid hydrogen fueled Centaur stage atop the Atlas predictably will double the payload afforded by conventional propellants. The Saturn V, being developed for a 90,000# lunar payload, uses liquid hydrogen in its upper stages, as do its predecessors Saturn I and IB.

This report, based upon a survey of the present day technical literature, discusses the value of using liquid hydrogen for high energy space boosters. The theorectical performance characteristics and parameters with liquid oxygen and liquid fluorine oxidizers are presented. Payload capabilities using this fuel are examined. Practical aspects of the advantages, disadvantages, and associated problems are explored when using liquid hydrogen fuel in conjunction with its required hardware and ground handling equipment. For a more complete appreciation of the significance of the specific performance characteristics a discussion of liquid rocket performance parameters is included.

This technical survey was accomplished and the present thesis was written in November and December, 1963 at the United States Naval Postgraduate School, Monterey, California.

2. LIQUID ROCKET PROPULSION PARAMETERS

General:

Rocket propulsion, a simple example of the reaction principle, may be defined as a means of locomotion in such a manner that a reaction is imparted to a vehicle by the momentum of ejected matter. The matter ejected is a high speed stream of gaseous particles usually generated by a chemical reaction between a fuel and oxidizer stored within the vehicle. In order to obtain the high velocity of the exhaust gas products they are accelerated through a convergent-divergent nozzle to supersonic speeds, thus completing the conversion of the thermal energy of a chemical reaction into kinetic energy of the products of combustion.

Total Thrust:

The total thrust imparted to a vehicle moving through a homogenous external fluid is composed of the momentum thrust and the pressure thrust. The former is the product of the mass flow rate of propellant and the exhaust gas velocity relative to the vehicle. The latter is the product of the exit area of the nozzle, or emergent stream, and the difference in pressure between this stream and the ambient pressure.

$$F = \frac{\dot{m}}{g} \quad V_e \quad + \quad (p_e - p_a) \quad A_e$$

The ideal isentropic flow through a convergent-divergent nozzle allows the full expansion of the exhaust gases to the ambient pressure. This condition gives a pressure thrust of zero and thus the optimum total thrust available. Under-expansion, $p_e > p_a$, indicates the nozzle exit area is not large enough to allow complete expansion of the working fluid; over-expansion, $p_e < p_a$, results in a shock condition in the nozzle divergent section. In either case there is a loss in total energy and a resultant decrease in available exhuast gas velocity.

Effective Exhuast Gas Velocity:

The effective exhaust gas velocity, V_{eff}, is defined as

$$V_{eff} = \frac{Fg}{\dot{m}} = V_e + (p_e - p_a) \frac{A_e g}{\dot{m}}$$

It is simply a measure of an effective velocity which generates the same total thrust obtained from the actual exhuast velocity and the pressure thrust. Its major importance lies in its use in defining more explicit performance parameters, and in evaluating test stand data when optimum expansion is not possible.

Specific Impulse:

Specific impulse is one of the most important parameters in evaluating rocket performance. Mathematically it is defined as

$$I_{sp} = \frac{F}{\dot{m}} = \frac{V_{eff}}{g}$$

Although much of the time specific impulse is given the dimension of seconds, it is actually a measure of thrust over propellant flow rate. More descriptively the proper units of I_{sp} are lbs (force)/lbs (mass)/sec. The numerical value of specific impulse can sometimes be misleading when comparing different types of propellant systems. It is dependent both upon total thrust and the mass flow rate of the working fluid. For example an electrical propulsion system might give a specific impulse several times greater than the highest value obtainable from a chemical system. However, because the mass of the particles being accelerated is microscopic in nature the resultant thrust is very small.

The relative importance of specific impulse is as a measure of the chemical and thermodynamic energy release available from the propellants themselves. I_{sp} is a property of the chemical composition of the propellant and can be theorectically evaluated by thermodynamic analysis. The following evaluation shows the dependence of the specific impulse on the thermodynamic conditions that exist in the combustion chamber and during the subsequent expansion process in the nozzle.

Applying Bernoullis energy equation per unit mass between the combustion chamber and nozzle exit gives the following.

$$\left[h_c + \frac{V_c^2}{2gJ} \right] - \left[h_e + \frac{V_e^2}{2gJ} \right] = q - W_o$$

The velocity, V_c, within the chamber is validly considered zero. Therefore, in an adiabatic process in which all work is accomplished by expansion the energy equation becomes

$$h_c - h_e = \frac{V_e^2}{2gJ}$$

or

$$V_e = (2gJ\Delta h)^{\frac{1}{2}}$$

For the case of optimum expansion (pressure thrust equals zero)

$$I_{sp} = \frac{V_e}{g} = \left(\frac{2J\Delta h}{g} \right)^{\frac{1}{2}}$$

Thus the specific impulse may be determined from the enthalpy change of the fluid in the system as it passes from the combustion chamber to the nozzle exit. Assuming a one dimensional isentropic expansion of an ideal gas in the rocket nozzle allows specific impulse to be further evaluated.

$$h = \int c_p \, dt$$

$$c_p = \frac{k}{k-1} \frac{R}{MW} \frac{1}{J}$$

therefore

$$\Delta h = \frac{k}{k-1} \frac{R}{MW} \frac{1}{J} (T_c - T_e)$$

isentropic conditions

$$T_e = T_c \left(\frac{Pe}{Pc} \right)^{\frac{k-1}{k}}$$

$$\Delta h = \frac{k}{k-1} \frac{RTc}{MW} \frac{1}{J} \left[1 - (pe/P_c)^{\frac{k-1}{k}} \right]$$

therefore

$$V_e = (2gJ\Delta h)^{\frac{1}{2}}$$

$$V_e = \left(\frac{2gRk}{k-1} \frac{Tc}{MW} \left[1 - (P_e/P_c)^{\frac{k-1}{k}} \right] \right)^{\frac{1}{2}}$$

and

$$Isp = \left(\frac{2R}{g} \frac{k}{k-1} \frac{Tc}{MW} \left[1 - (P_e/P_c)^{\frac{k-1}{k}} \right] \right)^{\frac{1}{2}}$$

The above equation shows that there are two important factors determining propellant performance or specific impulse. The first is the pressure ratio over which the propellant exhaust gases are expanded after combustion. For given ambient conditions this pressure ratio is dependent upon and limited by the physical size of the nozzle. The second factor is the propellant itself. Neglecting slight variation of specific heat ratio, k, and assuming a fixed set of hardware, specific impulse is directly dependent upon chamber temperature and molecular weight of the propellant.

$$I_{sp} \propto \left(\frac{T_c}{MW}\right)^{\frac{1}{2}}$$

A good propellant will give products with a high (T_c/MW) ratio.

If flame temperature, molecular wieght, and specific heat ratio could be chosen independently, really fantastic performance could be obtained. Unfortunately they are all rather intimately related in any one propellant so that attractive values of one of these properties are usually accompanied by less attractive or even detrimental values of the others.

In comparing the relative capabilities of propellant combinations one other form of specific impulse is sometimes referred to when applying these combinations to a specific unit of hardware. This additional parameter is called the volumetric impulse (ρI_{sp} - $\frac{lb_f\text{-Sec}}{ft^3}$) and takes into account the propellant density. The actual relationship between specific impulse and the respective propellant density is a complicated one, particularly for multistage missiles, and depends upon the specific missile application. It has, however, been shown that a change in specific impulse is always many times more significant on overall missile performance than a corresponding change in propellant density. Let it suffice here than to have mentioned this parameter and to know that it is little used in direct comparisons of specific propellants.

Characteristic Velocity & Thrust Coefficient:

These two parameters, characteristic velocity and thrust coefficient, are used to analyze respectively the combustion and expansion processes.

The combustion processes determine the ability of a propellant combination to generate useful energy whereas the expansion process determines the effectiveness of utilizing this energy.

By definition

$$C^* = \frac{P_c \, A^* \, g}{\dot{m}} \qquad \text{characteristic velocity}$$

$$C_F = \frac{F}{P_c \, A^*} \qquad \text{thrust coefficient}$$

The characteristic velocity, in essence, is a measure of the effectiveness with which the combustion products are generated in a particular rocket chamber. The thrust coefficient is sometimes looked upon as a nozzle efficiency.

Burnout Velocity:

The function of a rocket booster is to accelerate a payload to a prescribed velocity at a designated altitude. If at the time the fuel is exhausted the payload is traveling with the required speed in its intended direction, the rocket has done its job effectively. Thus the ultimate criteria of a rocket's performance are its velocity at the time the fuel burns out and the weight of the payload carried for a given rocket take-off weight.

These quantities are related by the general equation

$$V_B = V_e \ \ln\left(\frac{M_{LO}}{M_{BO}}\right)$$

or

$$V_B = I_{sp} \, g \, \ln\left(\frac{M_{LO}}{M_{BO}}\right)$$

This equation neglects factors such as initial velocity, drag and "g" losses. This relation is, however, not limited to single stage application. For multi-stage vehicles all upper stages plus the actual mission payload are considered as the effective payload of the first stage booster. A second stage will then assume an initial velocity equal to the burnout velocity of stage one, and the subsequent stages will be its effective payload. When rocket vehicles are staged in this manner, velocities achieved with each stage are additive, thus producing the very high velocities required for the final mission payload. Thus the basic design

of high performance rockets require that for each stage the ratio of lift-off weight to burnout weight (M_{LO}/M_{BO}) and the exhuast velocity (or I_{sp}) be as large as possible. Great strides are being made in the structural design of present day boosters; however, a practical limit has been reached in the value of M_{LO}/M_{BO} using todays state of the art materials. Thus, remembering the dependence of burnout velocity on specific impulse, it becomes obvious why attention has been focused upon increasing the performance available from the propellants used. An important point to note is the improvement in velocity that would be possible with improved propellant performance.

The approximate total velocity required for some typical space missions today are shown in Table I. Each figure represents the sum of each step of the respective mission, i.e. velocity required to escape the earth's pull of gravity plus that required to return from orbit around the moon.

TABLE I

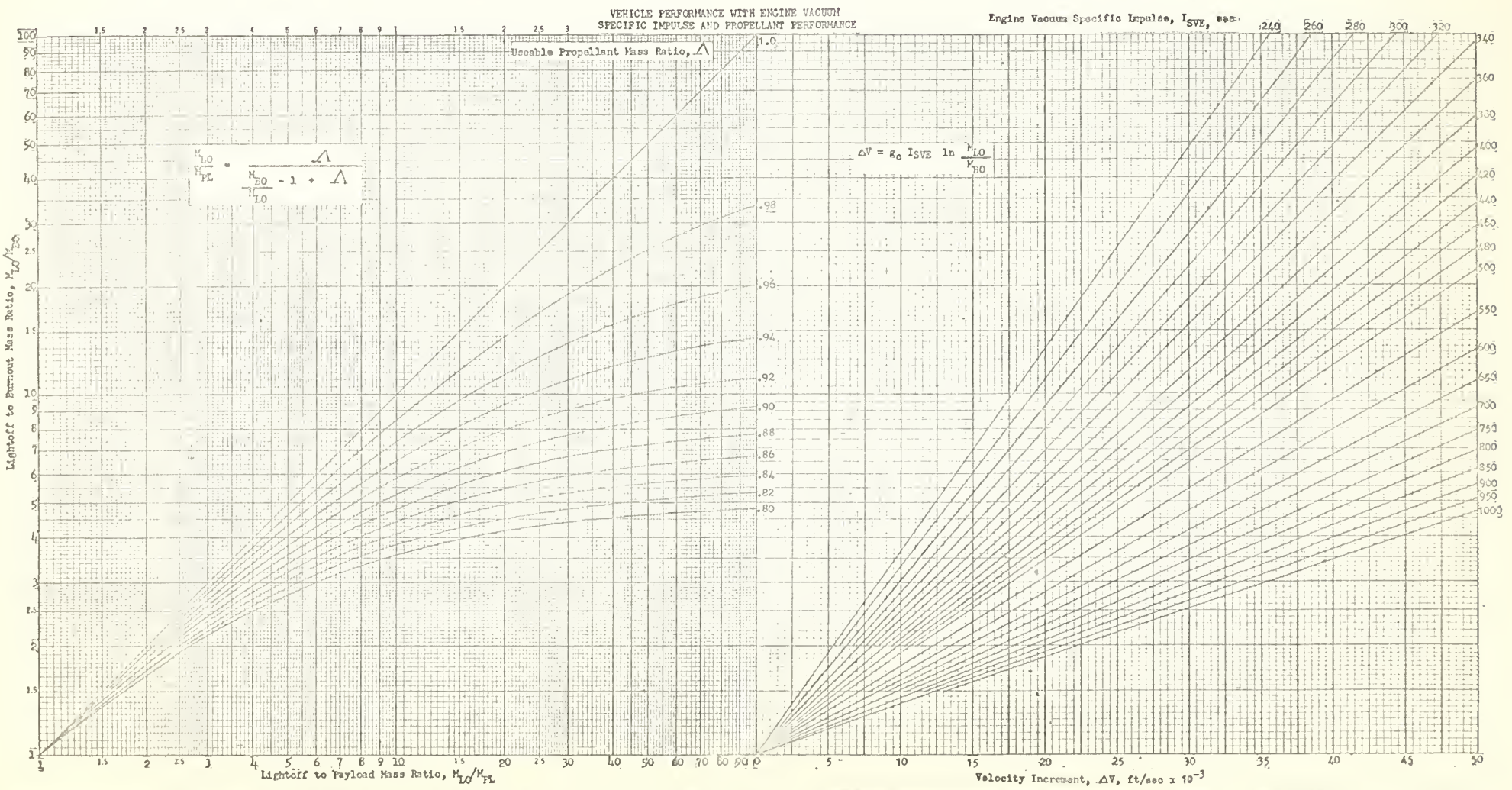
MISSION VELOCITY REQUIREMENTS

	mi/hr	ft/sec
ICBM	15,500	22,700
350 mile satellite	18,000	26,400
Moon satellite (or escape)	25,000	36,700
Circumnavigate moon & return	31,500	46,200
Earth to moon, land, take-off and return	42,500	62,400

Fig. 1 illustrates the general effect of increasing specific impulse upon the velocity increment per stage. The third important parameter, mass fraction, is the ratio of the weight of propellant to the total weight of the vehicle at lift-off. It is a physical result of vehicle hardware and requirements of design. This graph illustrates the importance of the chemical performance characteristics of the fuel in attaining a high specific impulse. Assuming a realistic lift-off to payload (M_{LO}/M_{PL}) mass ratio of five and a mass fraction (Λ) of .92, results in a velocity increment (ΔV) entirely dependent on the specific impulse.

FIG. 1

$$\Lambda = \frac{M_{PRO}}{M_{LO}} = 1 - \frac{M_{BO}}{M_{LO}}$$


VEHICLE PERFORMANCE WITH ENGINE VACUUM
SPECIFIC IMPULSE AND PROPELLANT PERFORMANCE
Useable Propellant Mass Ratio, Λ
Engine Vacuum Specific Impulse, I_SVE, sec.
$\frac{M_{LO}}{M_{PL}} = \frac{\Lambda}{\frac{M_{BO}}{M_{LO}} - 1 + \Lambda}$
$\Delta V = g_c I_{SVE} \ln \frac{M_{LO}}{M_{BO}}$
Lightoff to Burnout Mass Ratio, M_{LO}/M_{BO}
Lightoff to Payload Mass Ratio, M_{LO}/M_{PL}
Velocity Increment, ΔV, ft/sec x 10^{-3}

If this impulse were to vary from 280 to 380 lb-sec/lb, a conservative expectation of the high energy fuels, the velocity increment would increase approximately 30%.

3. PERFORMANCE CHARACTERISTICS OF LIQUID HYDROGEN

General:

The importance of gaining high specific impulse with chemical propellants arises principally from the fact that such propellants require tanks, a rocket thrust chamber and auxiliary equipment for operation. The weight of this equipment constitutes a lost payload, and sets an upper limit on the velocity increment which can be obtained with the first or any subsequent stages of a missile or space rocket. Since the ratio of this dead weight to propellant weight is approximately constant for any given propellant system, the velocity increase obtainable for any stage carrying useful payloads is limited to not much more than the exhaust velocity from the rocket engine nozzle. As has been shown this is simply the specific impulse times the acceleration of gravity.

A body must gain a velocity near 25,000 fps to orbit the earth as a satellite, and must go over 36,000 fps to escape the earth's gravitational field. Current exhaust velocities are about 9,000 fps, corresponding to a specific impulse of about 300 lb-sec/lb. This means three stages are required for satellites and four or more for escape missions unless payloads are decreased to infintesimal fractions of the total initial weight. An increase of specific impulse to a value of 400 lb-sec/lb would mean that satellites of two stages would be practical with a corresponding decrease in the number of stages for more ambitious targets. When it is realized that the useful payload, including total weight of any following stages, is generally not over ten per cent of the total initial lift-off weight, the importance of keeping the number of stages to a minimum is evident. Herein lies the requirement for high energy propellants.

Much research has been carried on in the past few years concerning these high energy propellants. However, few of these fuels and oxidizer combinations have been used in practical applications primarily because of the storage, handling, and safety problems involved. In many cases availability of the propellant itself, or of the structural hardware to

support such a high energy release continues to restrict the use of such fuels. It has only been recently that liquid hydrogen with its high available energy has stepped out of the ranks of the more exotic fuels. There are still problems in its use, but problems that are being contained so as to make liquid hydrogen the most promising rocket fuel presently available.

Perhaps the most immediate application of liquid hydrogen is in top stages of existing boosters to improve the space payload capability. For example, when a third stage is placed on an assumed conventional two stage vehicle the use of hydrogen in the third stage increases the payload delivered to a 300 mile earth orbit by ten per cent over that obtained with conventional propellants in the third stage. However, when even higher velocity requirements exist such as in a 24 hour orbit, the hydrogen third stage makes possible twice the payload afforded by conventional propellants in the same stage. Similarily, improvements in payload are possible if a hydrogen stage is used on top of present single stage vehicles.

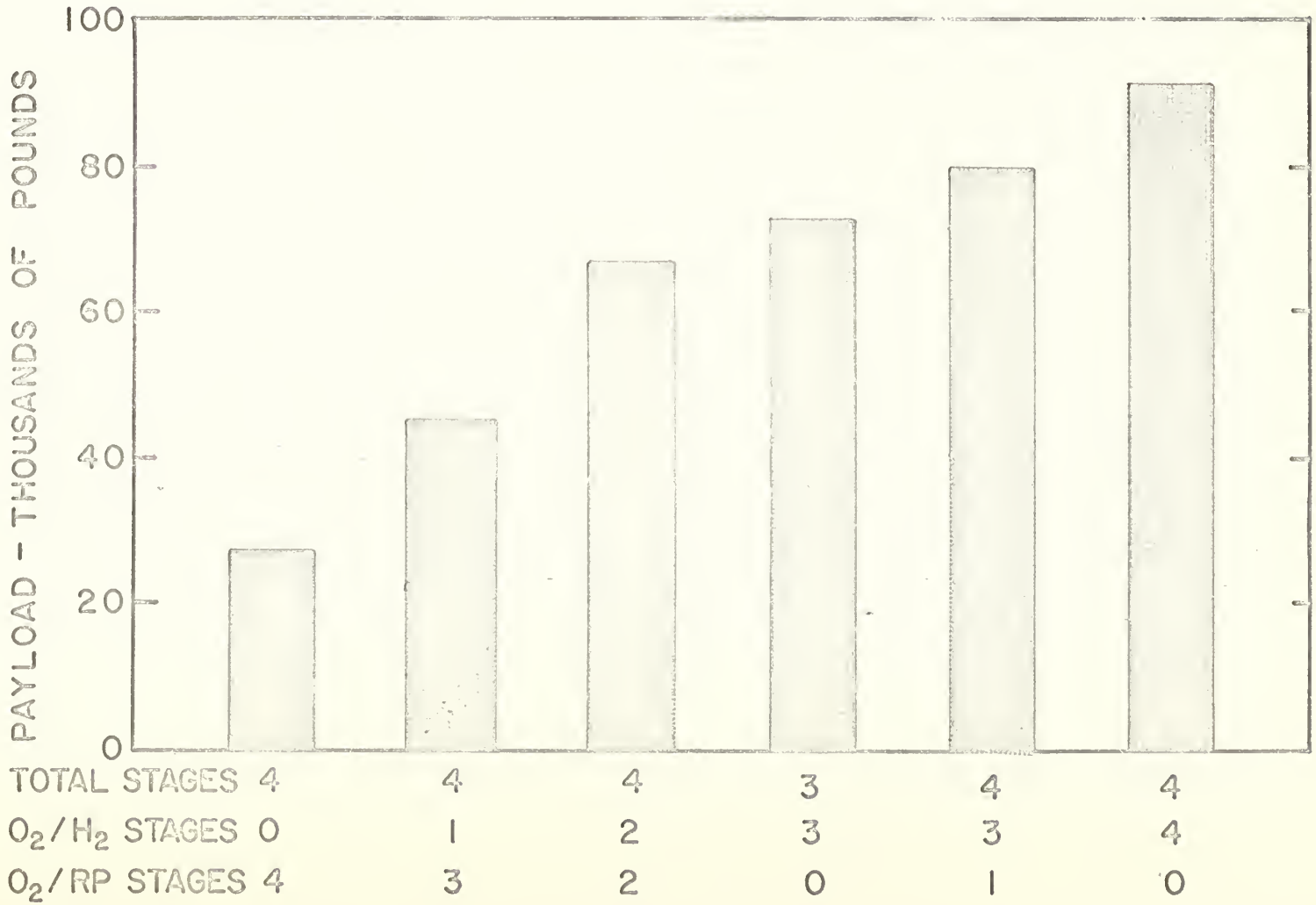
Although the payload of existing boosters can be improved dramatically by using hydrogen upper stages, significant improvements in payload, or reduction in size and number of stages, are also possible by using hydrogen in all stages of a multistage vehicle including the initial booster itself. The payload which a six-million pound thrust four stage vehicle can put into a 24-hour equatorial orbit increases from 25,000 lb, when all stages are oxygen-hydrocarbon, to 90,000 lb if all stages are oxygen-hydrogen. Fig. 2 emphasizes the payload gains possible by using high energy liquid hydrogen fuel in combination with liquid oxygen in one or more of the total stages.

These payload capabilities result from specific impulses almost 40 per cent higher than those offered by conventional hydrocarbon propellants. These increased payloads are possible in spite of the low density of hydrogen and the large tank volumes that are necessary for its storage.

Besides being desirable as a rocket propellant, primarily because of its chemical properties, hydrogen possesses the physical characteristics which make it an ideal working fluid in the nuclear rocket. Here again hydrogen is a standout with a specific impulse of well over twice that of any other expellant. At probable reactor operating temperatures,

FIGURE 2

24-Hr Equatorial Orbit Payload 6-Million-Lb-Thrust Booster



the specific impulse of nuclear rockets using hydrogen simply as the working fluid can be expected to double that of the best chemical systems.

Thermodynamic Performance:

The results of the theoretical performance calculations using liquid hydrogen as a rocket fuel depend primarily upon the combustion chamber pressure and the ratio of this pressure to the exhaust pressure that is used. The data that are presented here are all based upon calculations made at a combustion pressure of 1000 psia and exhaust pressures of 14.696 psia (1 atmosphere), 2.0 psia, and 0.2 psia. The latter exhaust conditions simulate high altitude operation where large nozzle area ratios are desirable for optimum thrust. The 1000 psia chamber pressure is not particularly descriptive of that required or even desired with liquid hydrogen, but it is rapidly becoming a reference pressure for liquid rocket performance comparisons.

Liquid hydrogen as the fuel in a particular propellant combination is theoretically compatible with many oxidizers. However, the two most important of these oxidizers are considered to be liquid oxygen and liquid fluorine; the former combination because it has present day application and plays a vital role in tomorrow's interplanetary boosters; the latter combination because it or its chemical derivatives offer a significant increase in performance over the LOX-LH2 system. For this reason the respective performance parameters discussed in conjunction with hydrogen are limited to the above two oxidizers. For reference purposes, however, Table II presents a summary of the maximum specific impulse obtainable from several propellant combinations all employing liquid hydrogen as fuel.

The maximum specific impulse is obtained from a shifting equilibrium condition of the exhaust gases as they flow through the nozzle. This condition releases the maximum energy available from the chemical reaction of the particular propellants. A frozen equilibrium condition is the theoretical minimum energy release. No recombination of the exhaust gases occurs in the nozzle; thus, there is no additional energy available. A more detailed analysis of the chemical energy release from combustion is presented in Appendix A.

Because of their significance, the following performance parameters are presented in a series of graphs in Fig. 3 thru 8 as a function of the propellant composition by weight.

Table II

Summary of Maximum Shifting Specific Impulse

Pc = 1000 psia
pe = 14.696 psia

Fuel	Oxidizer	Wt % Oxidizer	Shifting Isp, Seconds	Tc, °K
H_2	OF_2	85.5	411	3591
H_2	F_2	89	410	3964
H_2	O_2	78	391	2769
H_2	NF_3	93	351	3876
H_2	NO_2ClO_4	83	349	2713
H_2	ClO_3F	84	344	2744
H_2	N_2O_4	84	342	2660
H_2	H_2O_2	88	322	2404
H_2	ClF_3	92	318	3403
H_2	NH_4ClO_4	91	287	2448

Pc = 1000 psia
pe = 2.0 psia

Fuel	Oxidizer	Wt % Oxidizer	Shifting Isp, Seconds	Tc °K
H_2	OF_2	87	458	3756
H_2	F_2	92	457	4462
H_2	O_2	82	437	3182
H_2	NF_3	94.5	390	4212
H_2	NO_2ClO_4	86	389	3080
H_2	ClO_3F	87	383	3273
H_2	N_2O_4	86	381	2891
H_2	H_2O_2	91	359	2740
H_2	ClF_3	93.5	352	3690
H_2	NH_4ClO_4	92.5	319	2660

Pc = 1000 psia
pe = 0.2 psia

Fuel	Oxidizer	Wt % Oxidizer	Shifting Isp, Seconds	Tc °K
H2	OF_2	91	491	4149
H2	F_2	94	489	4809
H2	O_2	84	470	3374
H2	NO_2ClO_4	87.5	416	3259
H2	NF_3	94.5	413	4212
H2	ClO_3F	90	409	3502
H_2	N_2O_4	89	408	3254
H2	H_2O_2	92.5	385	2912
H2	ClF_3	95	373	3985
H_2	NH_4ClO_4	94	340	2883

1. Specific impulse (Frozen & Shifting)
2. Exhaust gas temperature
3. Chamber conditions (temperature, molecular weight, characteristic exhaust velocity)

Examination of these parameters produces complete information which permits analysis and comparison of the thermodynamic performance of liquid hydrogen. Referring to Fig. 3 it is seen that liquid hydrogen-liquid fluorine is an excellent propellant combination yielding a maximum theorectical I_{sp} of over 480 lb-sec/lb. It is interesting to note that the maximum performance occurs on the fuel rich side of stoichiometric combustion, indicating the value of hydrogen gas as a working fluid. In fact it has been found that dissociation is relatively slight on the fuel rich side of the peak performance; thus there is only a slight difference in the frozen and shifting specific impulse in this region. However, as the system becomes oxidizer rich, the combustion temperature gets higher and the degree of chamber dissociation increases rapidly. The spread between frozen and shifting performance increases accordingly. The prime energy source in this system is the formation of HF gas.

Liquid oxygen rates closely behind liquid fluorine as an energetic oxidizer for liquid hydrogen. Again the maximum performance occurs on the fuel rich side of stoichiometric combustion. The principal source of energy in this system is obviously water. Dissociation and the corresponding spread between shifting and frozen specific impulse increase as the system becomes oxidizer rich. The most significant difference between the H_2-O_2 and the H_2-F_2 systems is the lower combustion temperature of the former. Although somewhat higher performance may be achieved by oxidizing hydrogen to hydrogen-fluoride rather than water, a considerably hotter flame must be tolerated.

It is obvious from the comparison of the peak impulse conditions for shifting equilibrium flow of the two systems that hydrogen-fluorine not only yields a higher performance, but it does so at about one-half of the hydrogen weight flow required by the hydrogen-oxygen combustion. As a consequence of these factors there result considerable reductions in tank volume and hence structural weight. However, it is also evident that the specific impulse increment between frozen and shifting equilibrium conditions for hydrogen-fluorine is disturbingly large. In fact,

FIGURE 3

LIQUID HYDROGEN-LIQUID FLUORINE PERFORMANCE CURVES

pc = 1000 psia

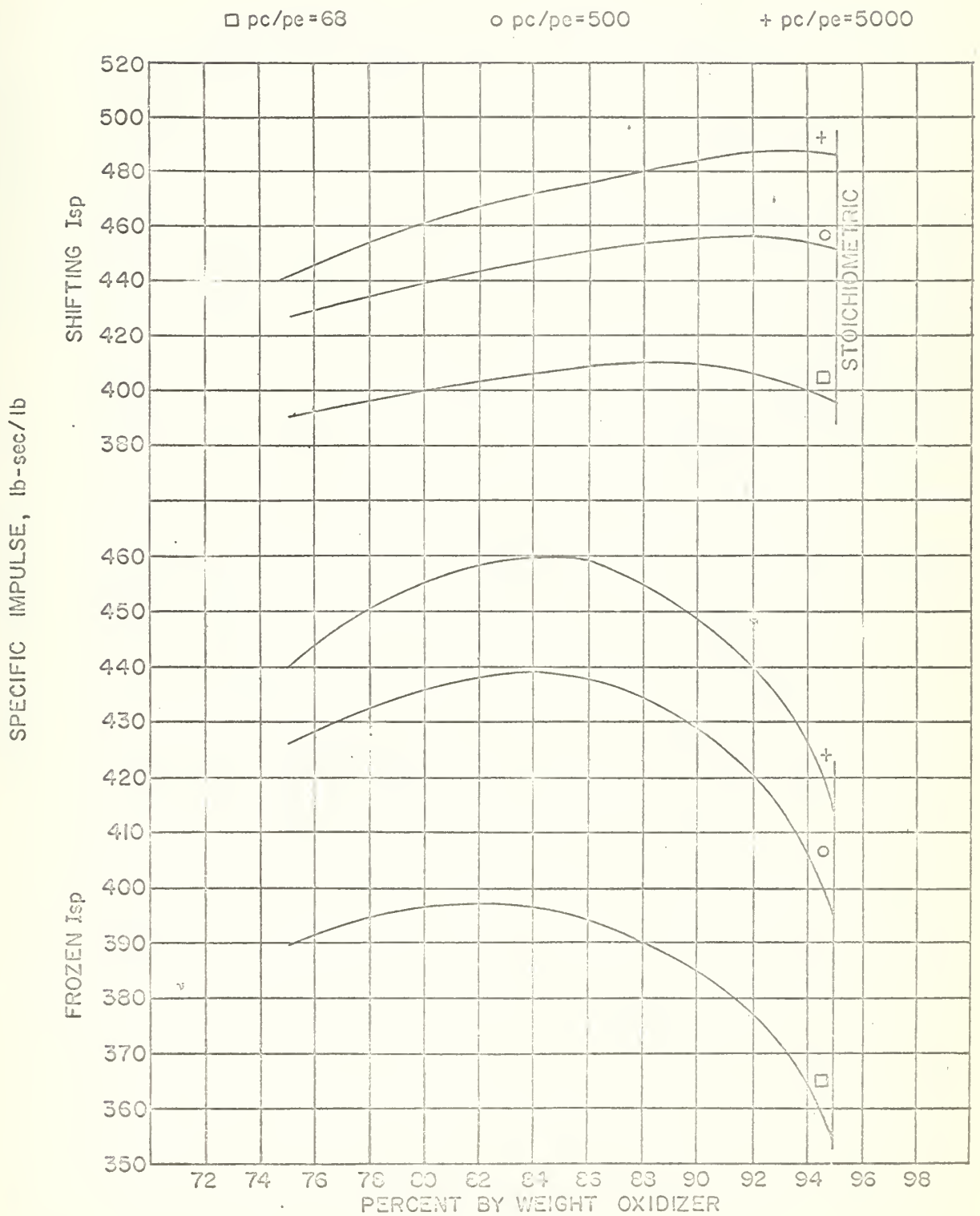


FIGURE 4

LIQUID HYDROGEN-LIQUID FLUORINE PERFORMANCE CURVES

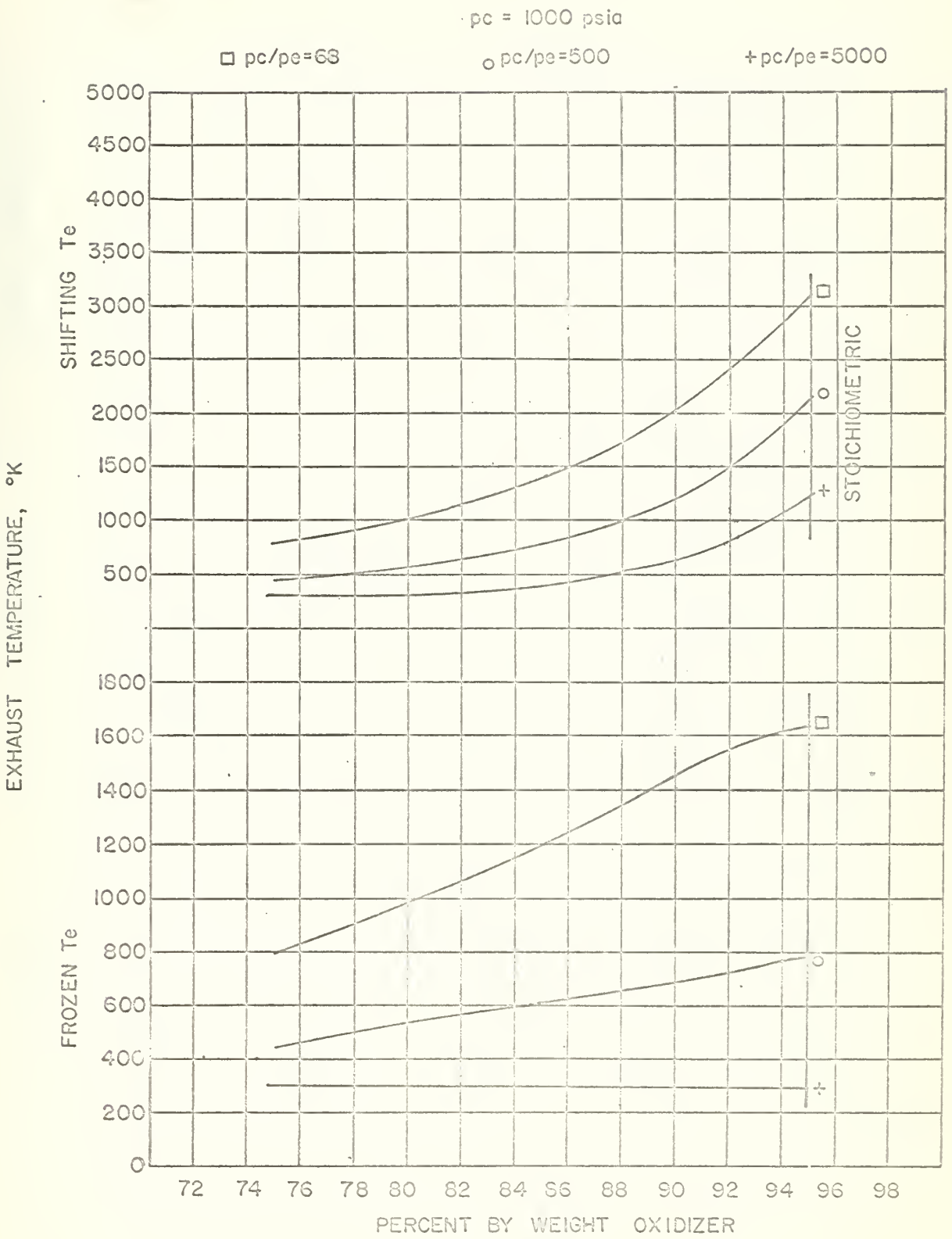


FIGURE 5

LIQUID HYDROGEN-LIQUID FLUORINE PERFORMANCE CURVES

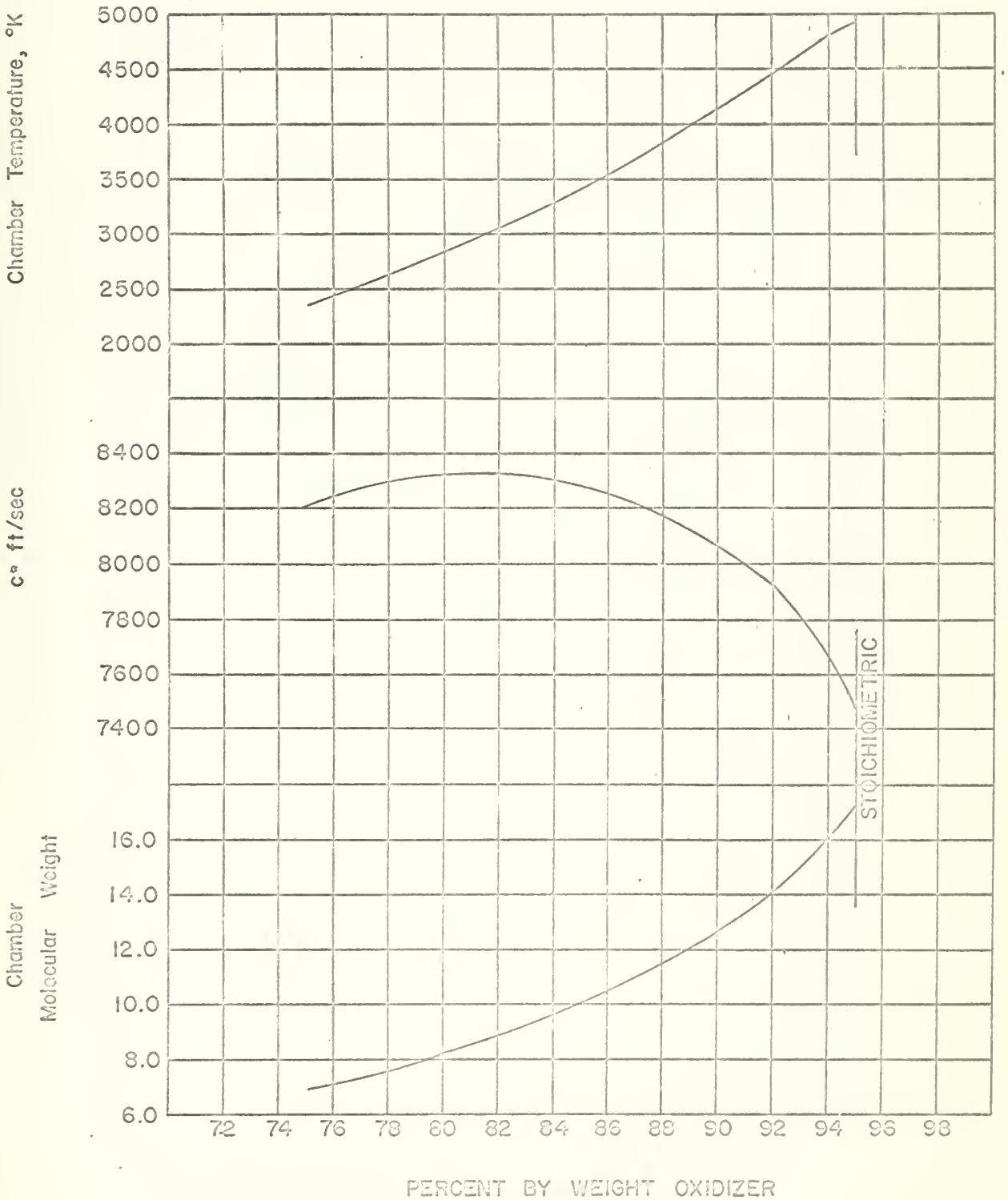


FIGURE 6

LIQUID HYDROGEN - LIQUID OXYGEN PERFORMANCE CURVES

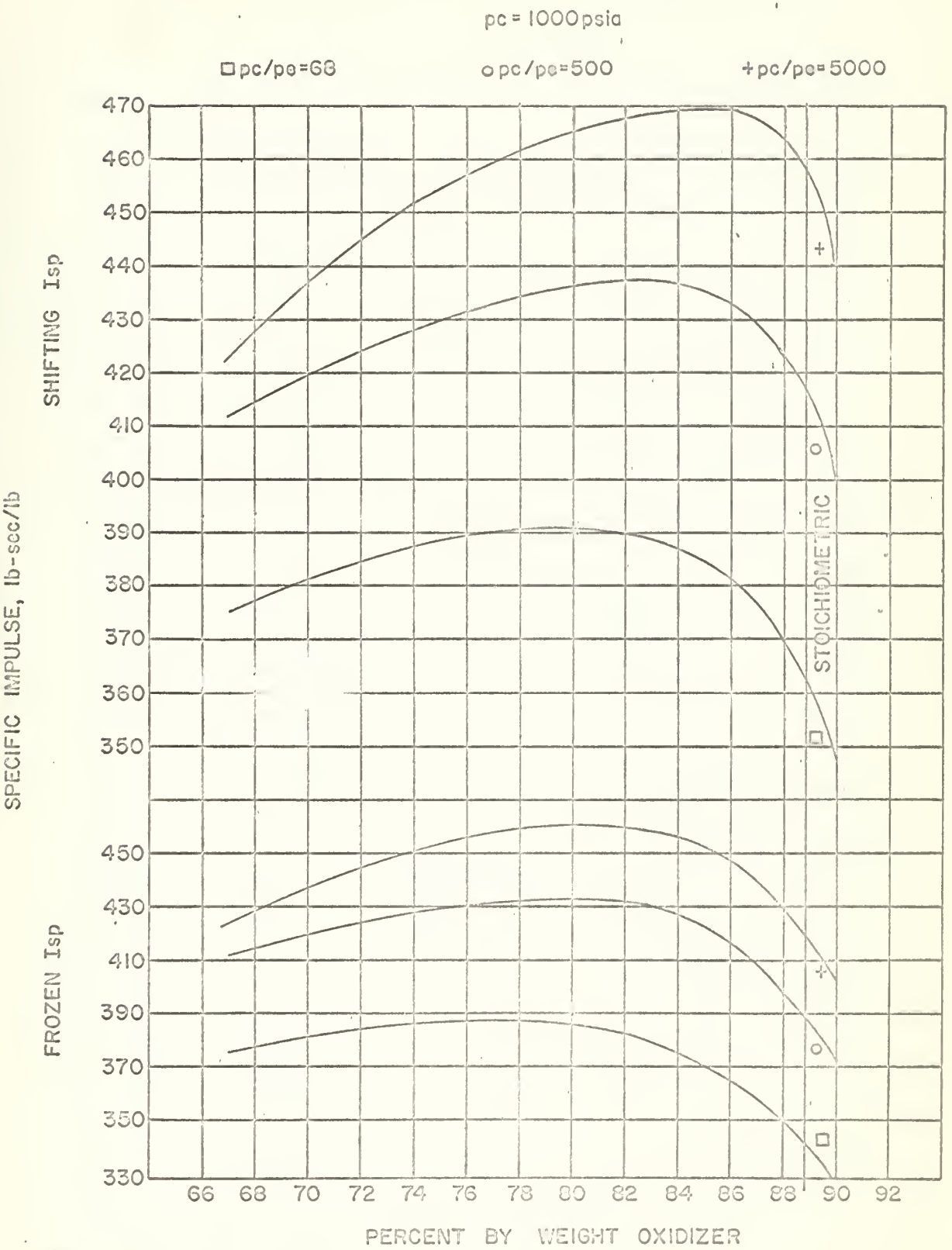


FIGURE 7

LIQUID HYDROGEN-LIQUID OXYGEN PERFORMANCE CURVES

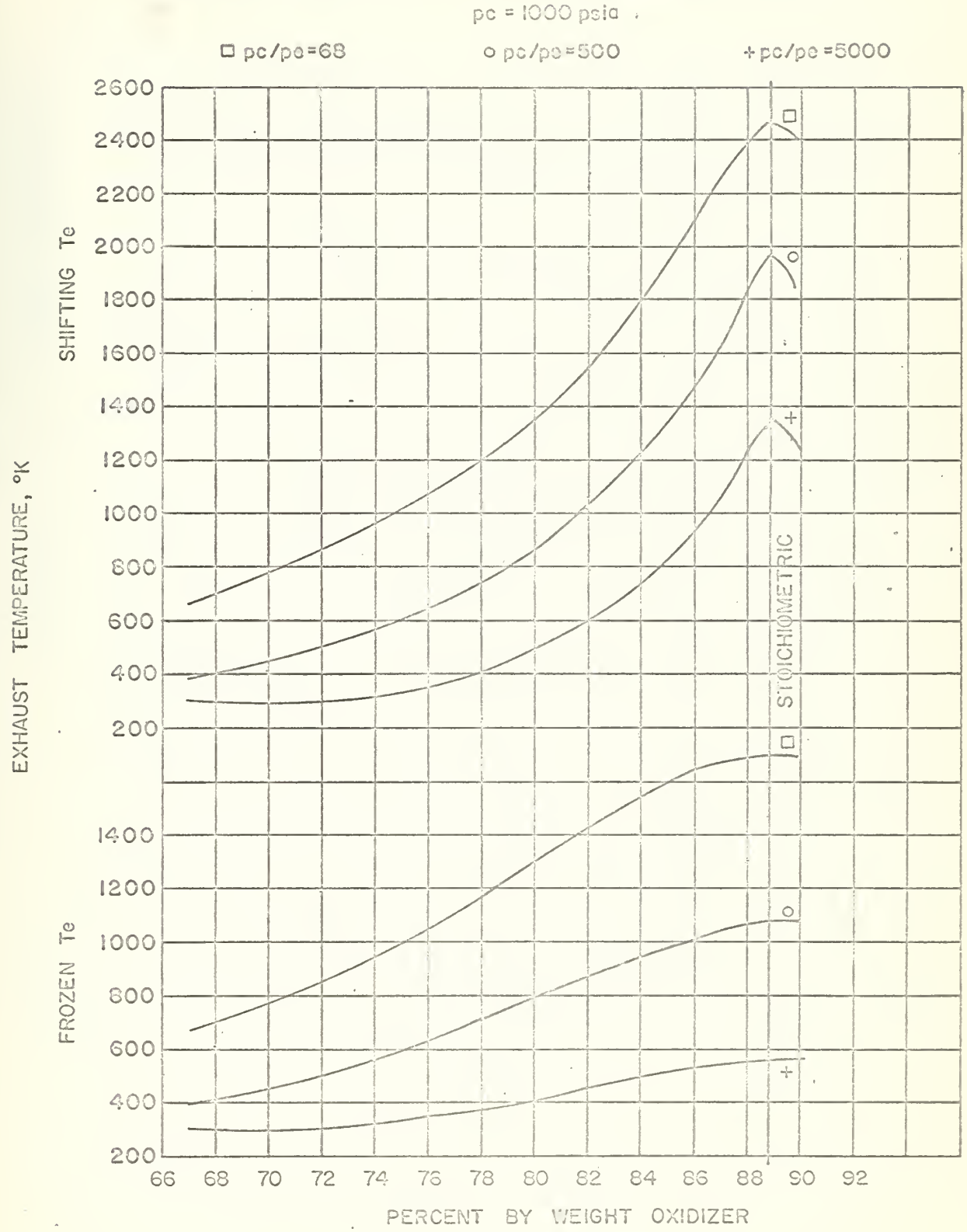
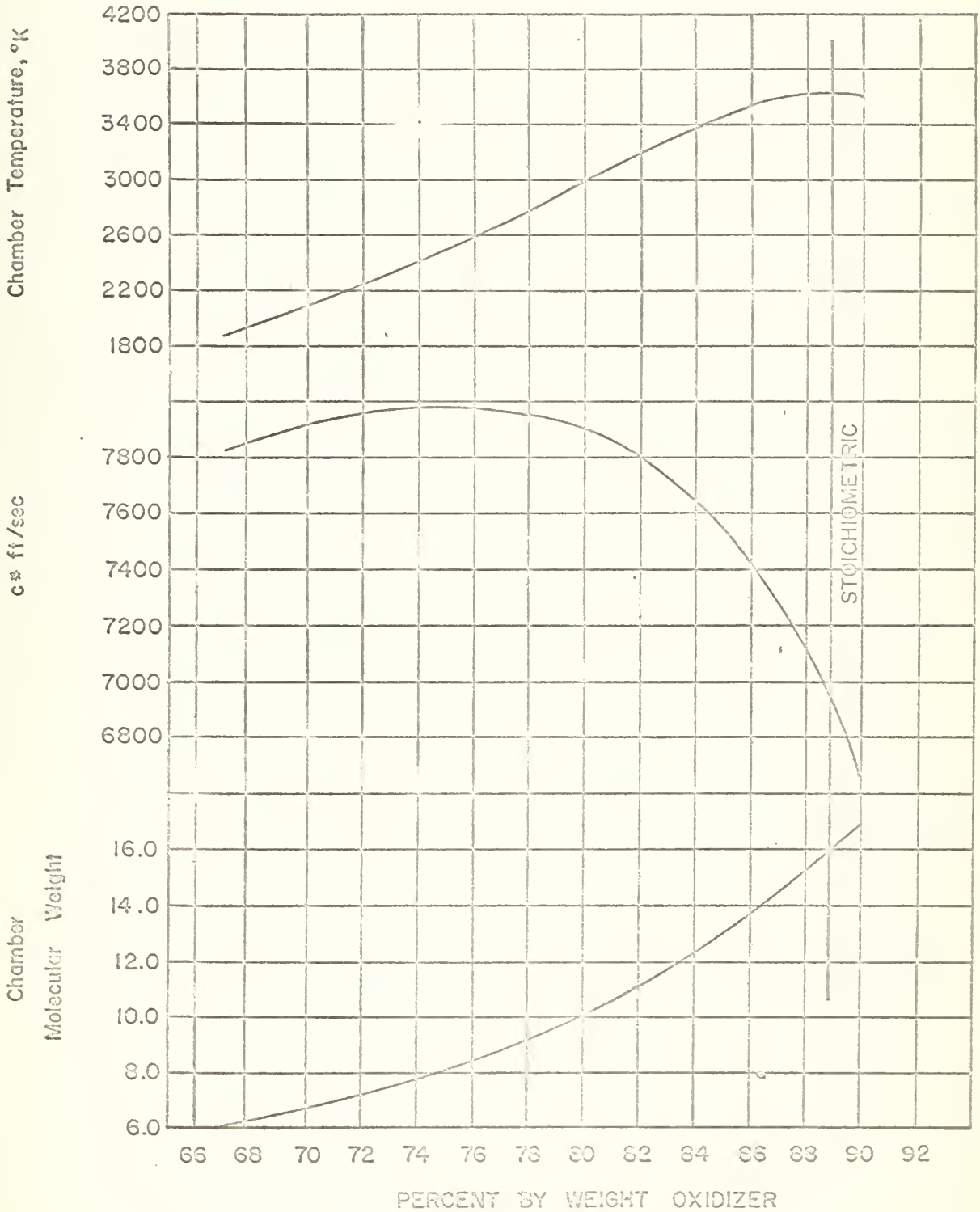


FIGURE 8

LIQUID HYDROGEN-LIQUID OXYGEN PERFORMANCE CURVES



at peak impulse it is about three times as great as for hydrogen-oxygen. This increment in specific impulse, which represents the dissociation energy, becomes larger as chamber pressure is reduced and/or nozzle area ratio increases (P_c/p_e increases), and implies that unless near equilibrium performance can be obtained the advantages of hydrogen-fluorine over hydrogen-oxygen are lost. Furthermore, the peak impulse mixture ratio shifts toward the stoichiometric value as the pressure ratio (P_c/p_e) increases (as is the case for upper stage applications) with corresponding increases in combustion gas temperature. This, of course, poses a severe regenerative cooling problem since the flow rate of liquid hydrogen available for cooling is reduced.

Fig. 9 is a graph comparing the maximum specific impulse available at a pressure ratio of 68 (P_c = 1000 psia, p_e = 14.696 psia) of the liquid hydrogen combinations just previously discussed, and two more widely used and proved propellants.

Hydrazine fuel (N_2H_2) in combination with nitrogen tetroxide oxidizer (N_2O_4) offers a good propellant for comparing the relative merits of liquid hydrogen. This system, or slight modifications thereof, is presently being used in operational military ballistic missiles because of its long term storability characteristics. In addition, this propellant combination is planned for space propulsion, not only because it is storable indefinitely, but also because it has a high reliability restart feature.

Polyethylene, with the empirical formula $(CH_2)_X$, was chosen as the prototype fuel to represent the hydrocarbon family. Having the highest O/C ratio of any of the hydrocarbons, CH_2 represents the maximum energy source from the family. Hydrocarbons in conjunction with liquid oxygen have a long history as a rocket fuel, and they still play a vital role in both the military and space effort. RP-1 and LOX are destined to provide the energy required for the 7½ million pound first stage of Saturn V.

Purely from a chemical performance viewpoint, it is easy to see the advantage of liquid hydrogen as a fuel. The total energy release produces over 1½ times the specific impulse than that available from either hydrazine or the hydrocarbons.

When examining Fig. 10, which compares the chamber temperature of the same four propellants, it is obvious that hydrogen combinations

FIGURE 9

PROPELLANT PERFORMANCE CURVES

pc = 1000 psia pe = 14.696 psia

pc/pe = 68

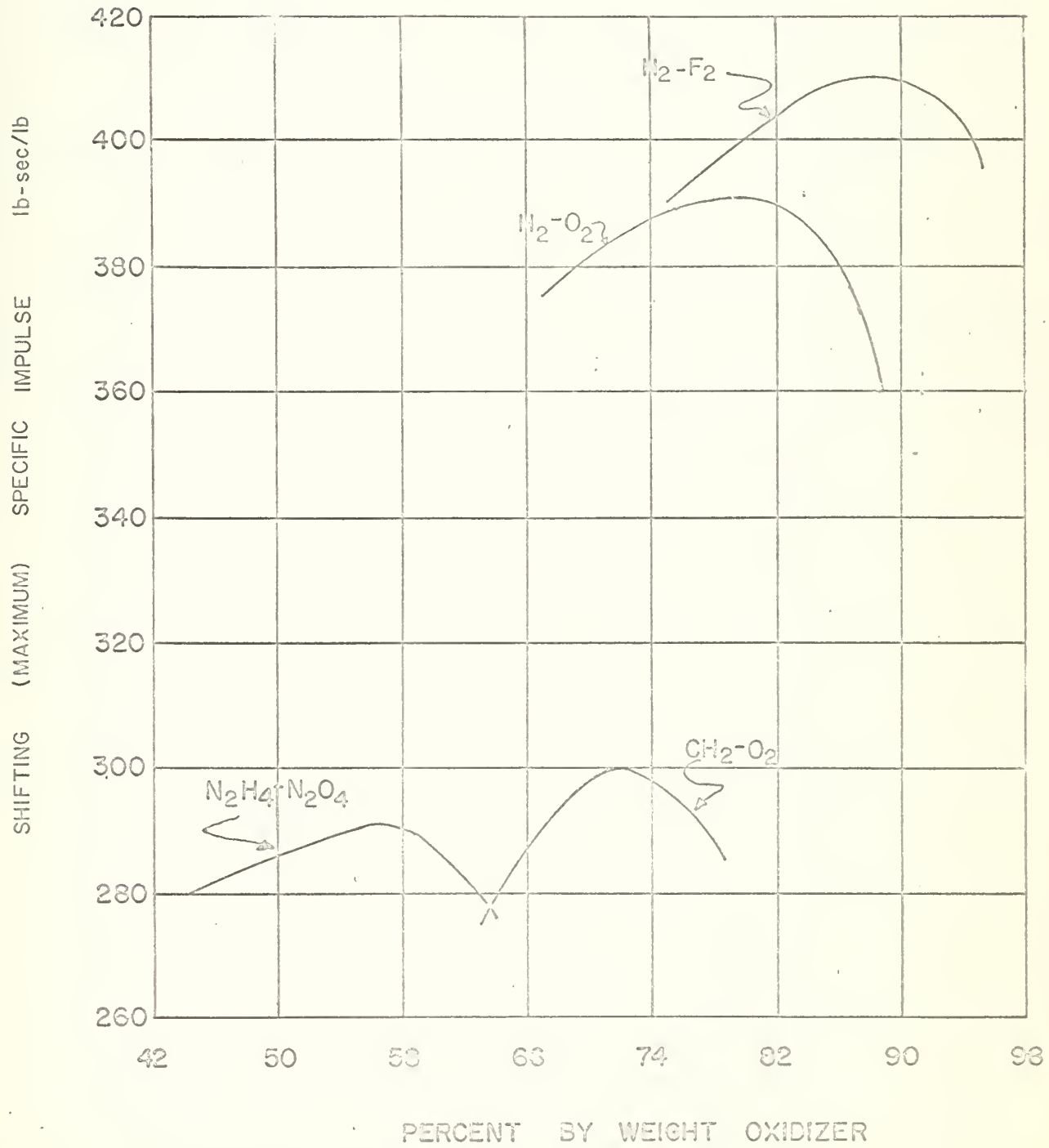
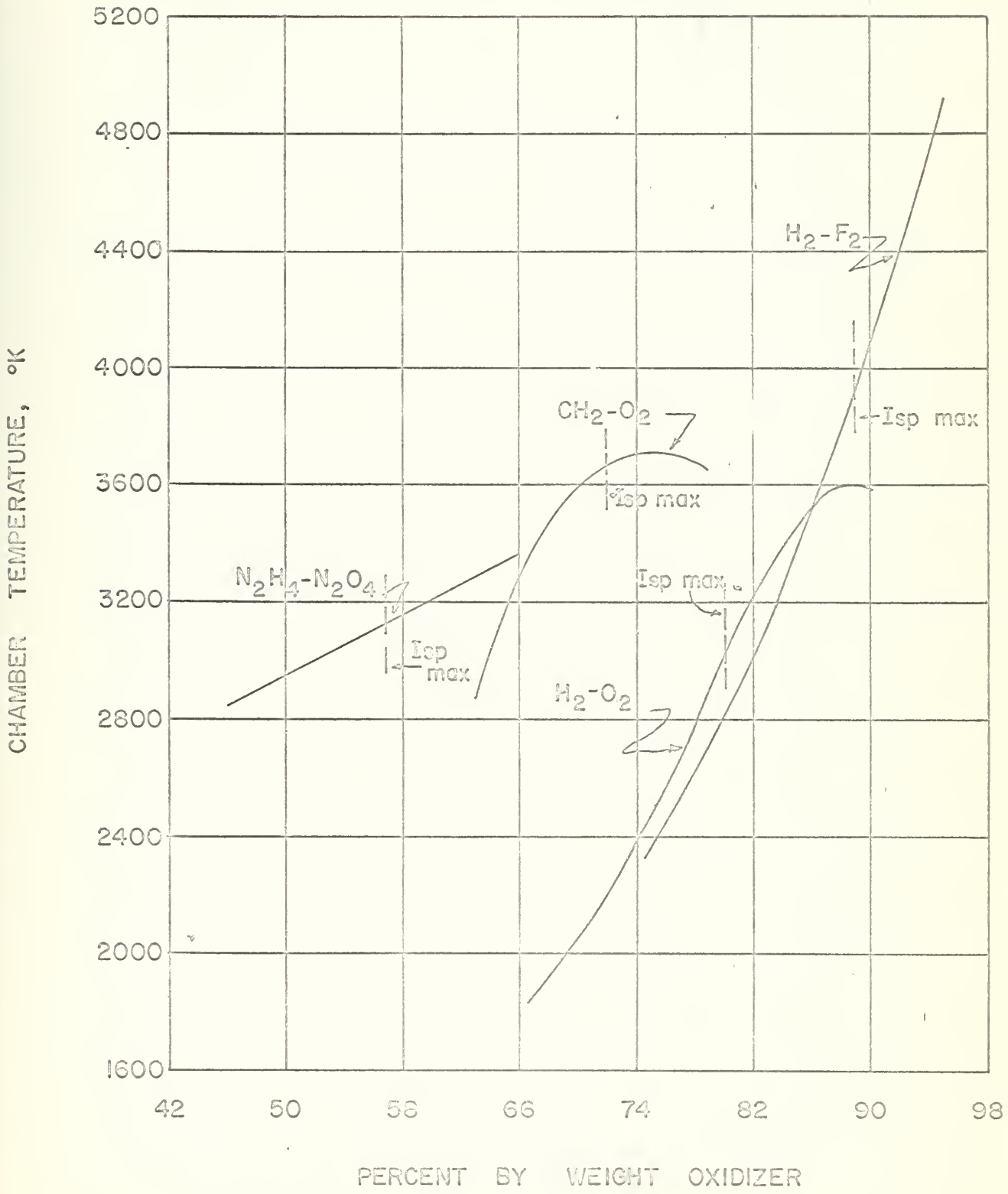


FIGURE 10

PROPELLANT PERFORMANCE CURVES



introduce no critical high temperature problems. Whereas the H_2-F_2 system produces the highest combustion temperature at the maximum performance point, the H_2-O_2 combination produces the lowest. Of great significance in this temperature comparison is the fact that liquid hydrogen is one of the best regenerative coolants available in rocket propellants.

Payload Capability:

In Fig. 11 is presented the payload capability of 65,000 lb. thrust hydrogen-oxygen and hydrogen-fluorine engines. This payload capability, defined as actual payload fraction of the total lift-off weight, is plotted as a function of the ideal velocity increment required. For a more complete grasp of the significance of the relative advantages of liquid hydrogen, similar curves are shown for LOX-RP1 and nuclear rocket engines. It is easy to see that the hydrogen engines rank well above the more conventionally used propellants. Significant increase in the payload fraction is realized in the nuclear engines both manned and unmanned. Both of these nuclear engines use hydrogen as the working fluid. The principle difference between the two nuclear rocket propulsion systems is the amount of shielding required for manned flight. At the present time the weight requirements are so high for the shielded engine that the question arises as to whether the nuclear rocket engine possesses enough of an energy increase over the hydrogen systems to make it worthwhile for manned flights. The obvious answer, which will realize the great potential of the nuclear engine and far surpass its chemical predecessors in payload performance, is to produce a light weight, effective radiation shield. Until that time chemical hydrogen systems represent the most attractive method of rocket propulsion. The right-hand scale on Fig. 11 shows the actual increase in payload of the nuclear systems over the H_2-F_2 systems.

Besides expressing the payload capability simply as a function of a general parameter like velocity increment, it can be specified for a particular mission analysis. The results of one such analysis are shown in Fig. 12 for a trip from earth orbit to just reaching Mars. The analysis assumes that a 9,000 lb. vehicle is put into earth orbit by the

FIGURE II

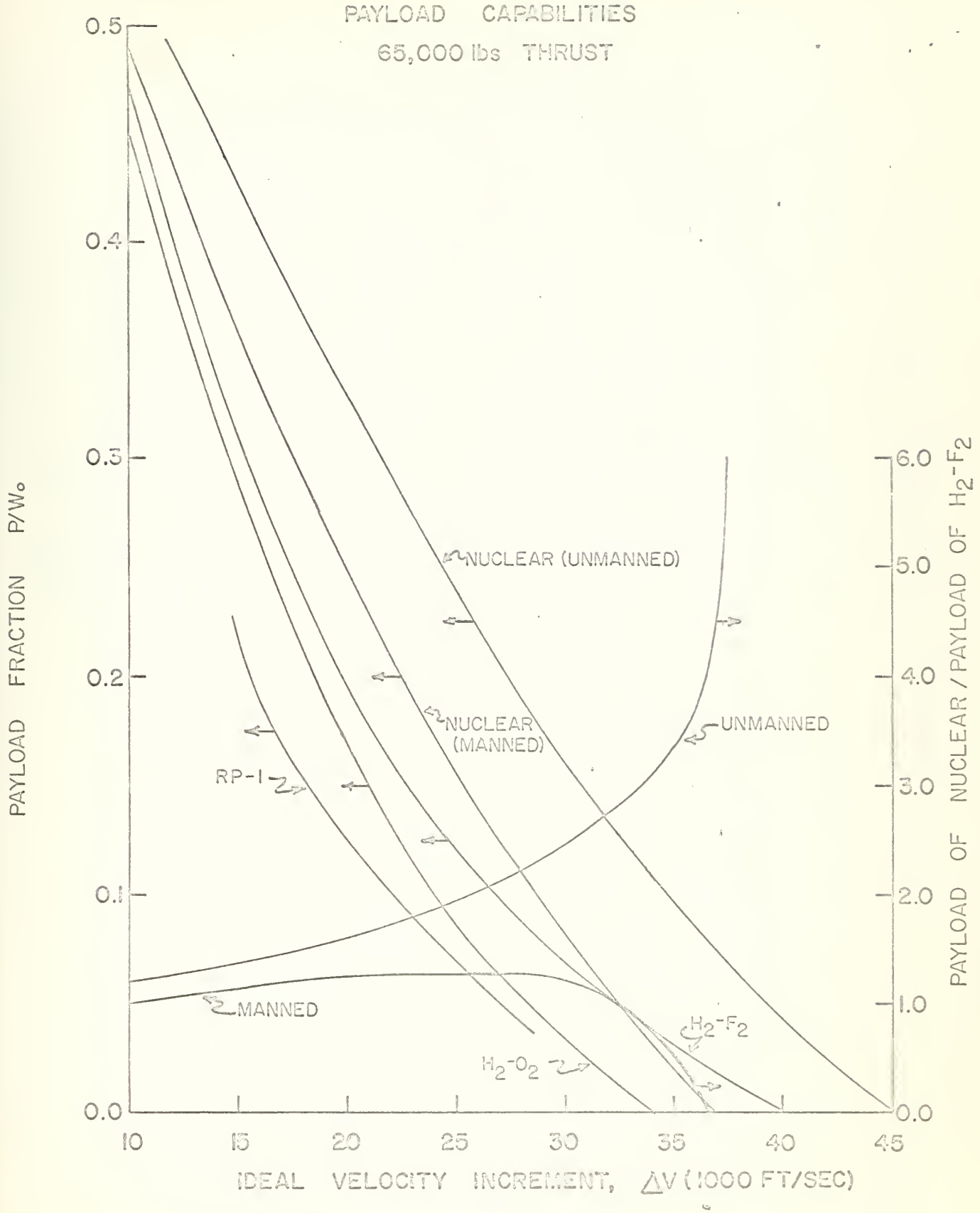
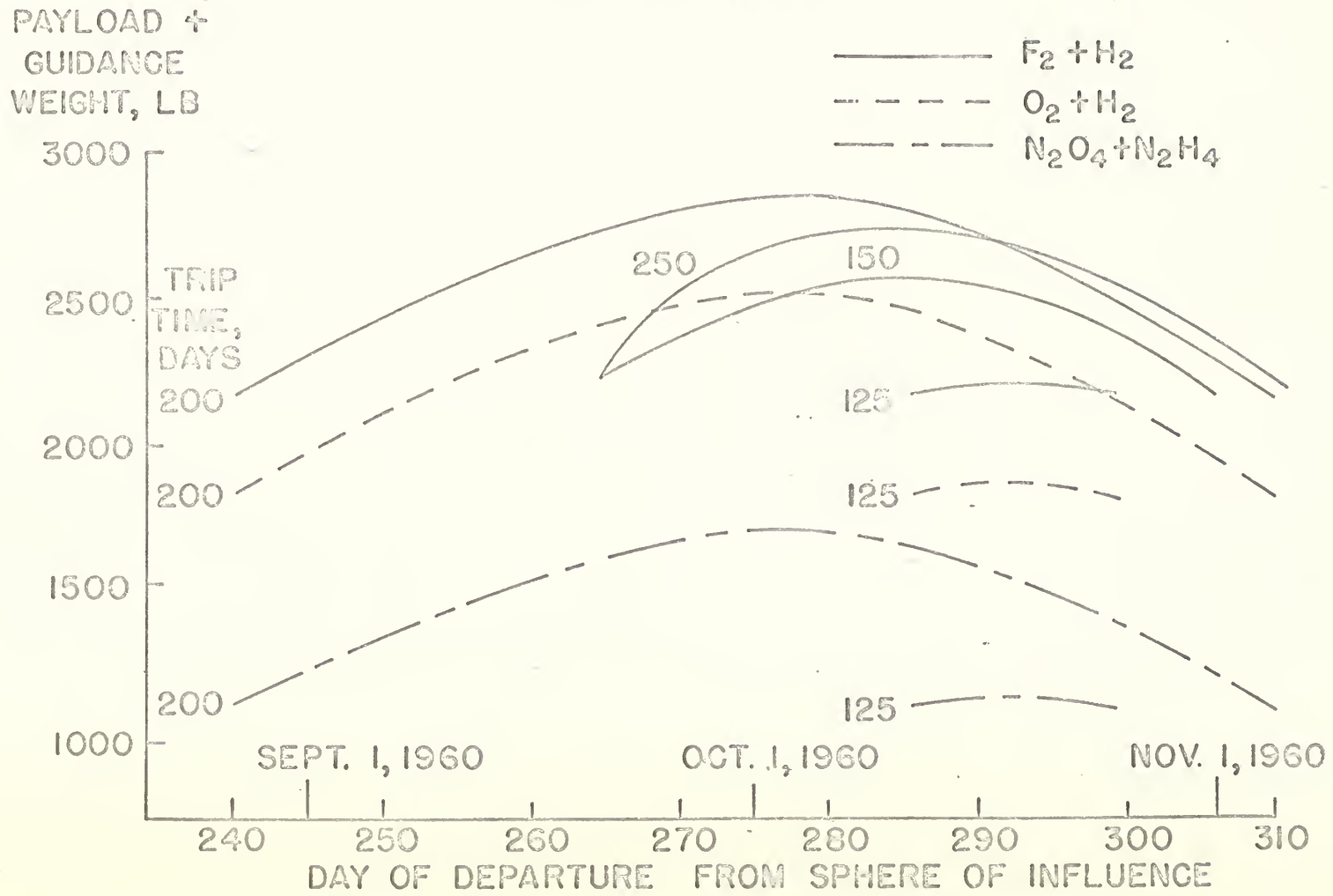

PAYLOAD CAPABILITIES
65,000 lbs THRUST
PAYLOAD FRACTION P/Wo
PAYLOAD OF NUCLEAR / PAYLOAD OF H2-F2
0.5
0.4
0.3
0.2
0.1
0.0
6.0
5.0
4.0
3.0
2.0
1.0
0.0
NUCLEAR (UNMANNED)
NUCLEAR (MANNED)
UNMANNED
RP-1
MANNED
H2-F2
H2-O2
10
15
20
25
30
35
40
45
IDEAL VELOCITY INCREMENT, ΔV (1000 FT/SEC)

FIGURE 12

SYSTEM PERFORMANCE, MARS PROBES

ATLAS-CENTAUR BOOSTER
9000 LB STAGES



Atlas-Centaur, a system comprising the Atlas topped with a hydrogen-oxygen second stage. If this 9,000 lb. vehicle is powered with any one of the three propellants shown, the payload plus guidance plotted in Fig. 12 can be given a 10,800 miles/hr velocity increment, 15% more energy than the bare minimum needed to reach Mars. This velocity increment permits choices among take-off date, trip time, and payload as shown. Hydrazine-nitrogen tetroxide system performance is shown, again as being illustrative of current storable systems, and in this analysis performs almost identically with the RP1-LOX system. This analysis incorporates sizes, weights, and engine performance commensurate with the respective propellants.

The distinct advantage of using high energy propellants based upon hydrogen is plainly illustrated. An analysis of even larger mission requirements, increased payloads projected deeper into space, would show these advantages over conventional propellants to be even larger.

4. PRACTICAL APPLICATION OF LIQUID HYDROGEN

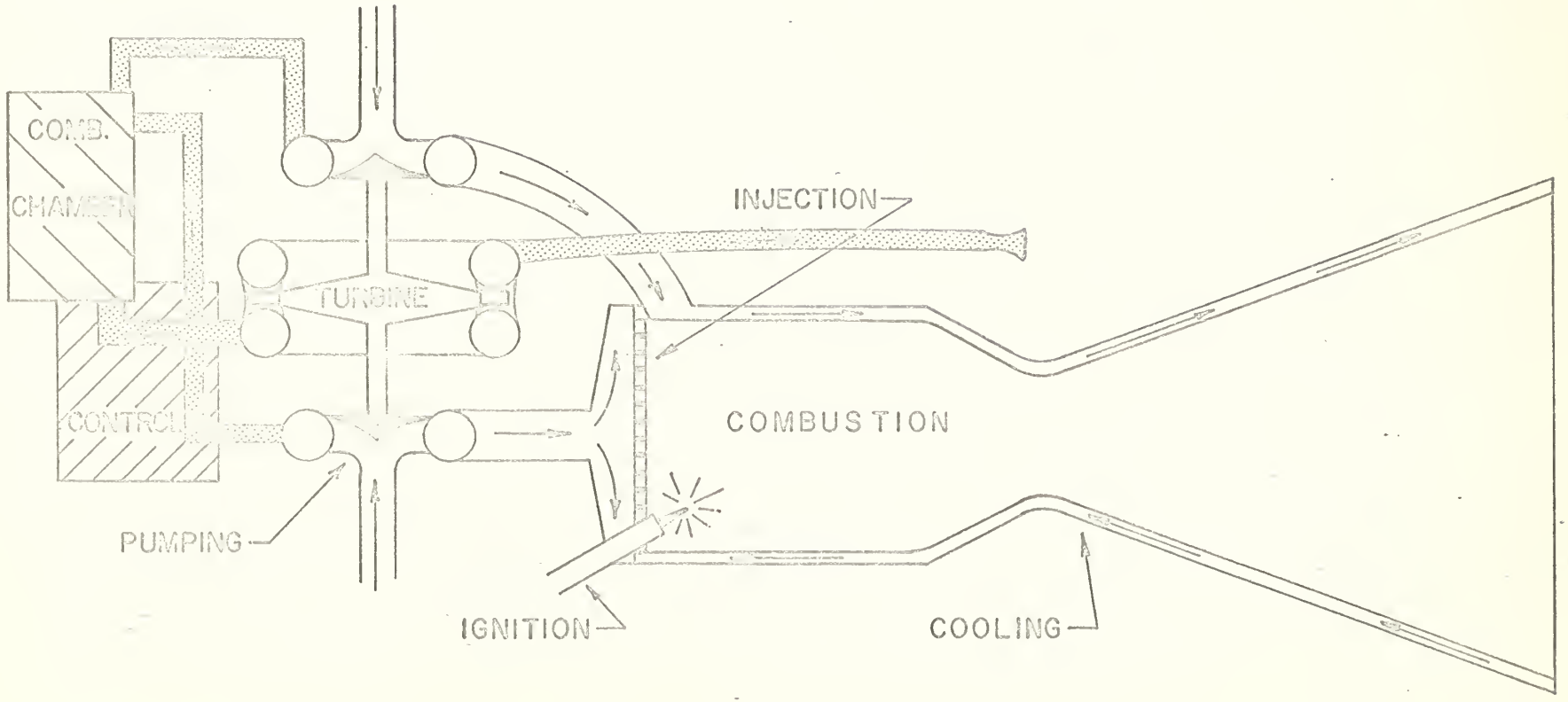
General:

Up to this point the discussion of the relative merit of using liquid hydrogen as a high energy propellant has considered only the theorectical performance of this fuel. Although this performance is attractively high, as with other rocket propellants there are advantages, disadvantages, and problem areas associated with the use of hydrogen. In order to effectively evaluate any propellant system beyond that of its calculated or laboratory performance, several additional areas have to be examined. Without operable subsystems, ground handling and storage compatibility, and a reasonable level of associated safety involved, even the most energetic of propellants is useless.

What is a problem area for the conventional propellants might very well be a distinct advantage for liquid hydrogen; however, the inverse might also be quite true. As a guide to the discussion which follows, Fig. 13 illustrates liquid rocket engine problem areas. The main problem areas in liquid rockets are, of course, the required tanks, the fluid flow systems, and the turbo pump units. Controls are probably not

FIGURE 13

ROCKET ENGINE PROBLEM AREAS



more affected by high energy propellant combinations than by any others; provided the fluid metering can be managed. Ignition and combustion must be considered Cooling is a problem because of the high combustion temperature that might prevail. These areas plus those of additional importance which affect the practical application of liquid hydrogen as the fuel in rocket engines are disucssed.

Tanks and Fluid Systems:

In considering the use of liquid hydrogen as a high energy space age fuel special attention must be given to the effect of low propellant bulk density and low temperature storage requirements on the design and size of the propellant tanks.

The low propellant bulk density associated with propellant combinations using liquid hydrogen as the fuel is one of the primary disadvantages of these systems. The volume of the necessary tanks depends on the mixture ratio of fuel and oxidizer at the desired maximum performance, and on the oxidizer used. Because of the low specific gravity of liquid hydrogen, the resultant tankage is comparatively large in all cases. Fig. 14 is a comparison sketch of three such tankage systems that might be required. Respective tank volumes are shown for liquid hydrogen in combination with liquid oxygen and with liquid fluorine. The tankage shown for the hydrazine-nitrogen tetroxide is an average representation of present conventional propellants. To carry the same total weight of propellant, the volume of the tanks for the conventional propellants need to be only 19.4% of that for a comparable LOX-LH2 stage. The tanks for the fluorine-hydrogen stage need only be about 50% of the total volume required for the LOX-LH2 system. It is obvious that the bulk density of the propellants has an important bearing on the weight of the tanks and therefore influences the payload performance capability of any particular system.

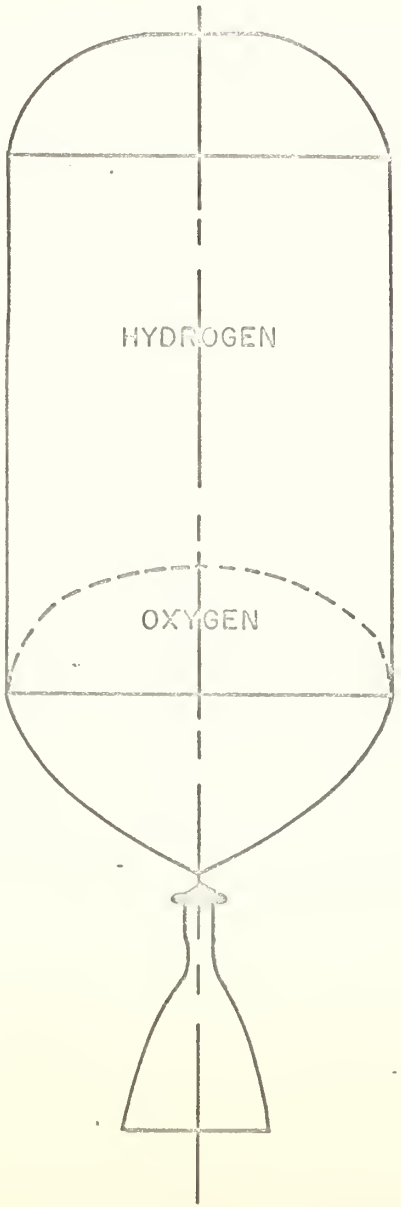
Appendix D contains plots of the specific gravity and the density of liquid hydrogen, and additional propellants of interest, for particular values of temperature within their respective required storage range. At its boiling point (-252.80C) liquid hydrogen has a density of about 4.44 lbs/cu.ft This contrasts markedly with hydrazine and RP-1 which

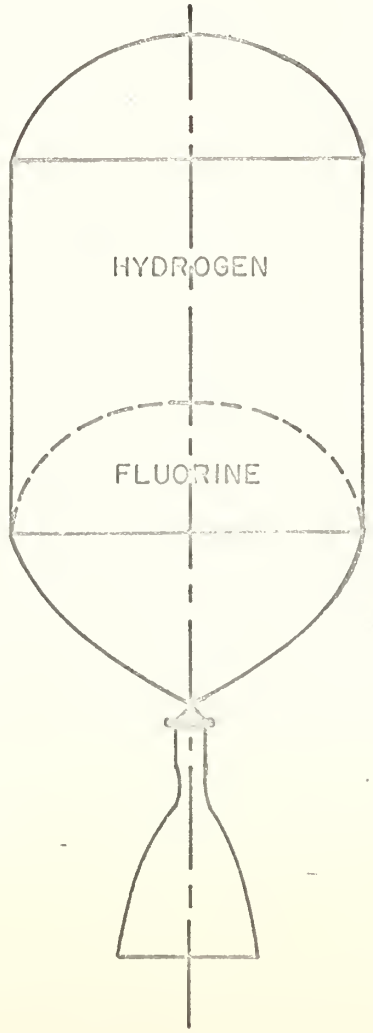
FIGURE 14

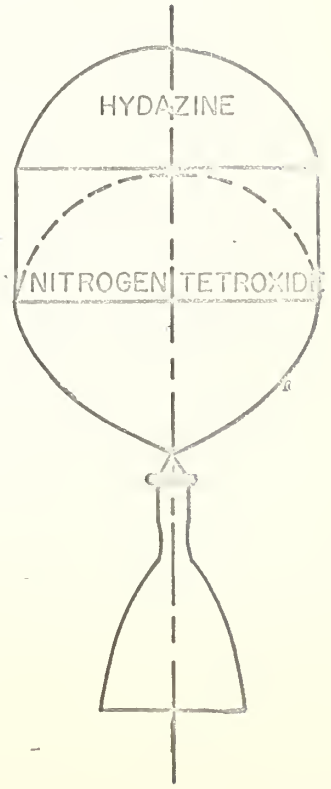
PROPELLANT TANK

BULK DENSITY

	H_2-O_2	F_2-O_2	N_2H_4-N_2O_4
MR O/F	2.89	9.42	1.0
BULK DEN	14.4 lb/ft^3	28.5 lb/ft^3	74.5 lb/ft^3
VOLUME	1.0 REF	.504	.194







have densities of 63.0 and 49.8 lbs/cu. ft. respectively at a storage temperature of 15°C.

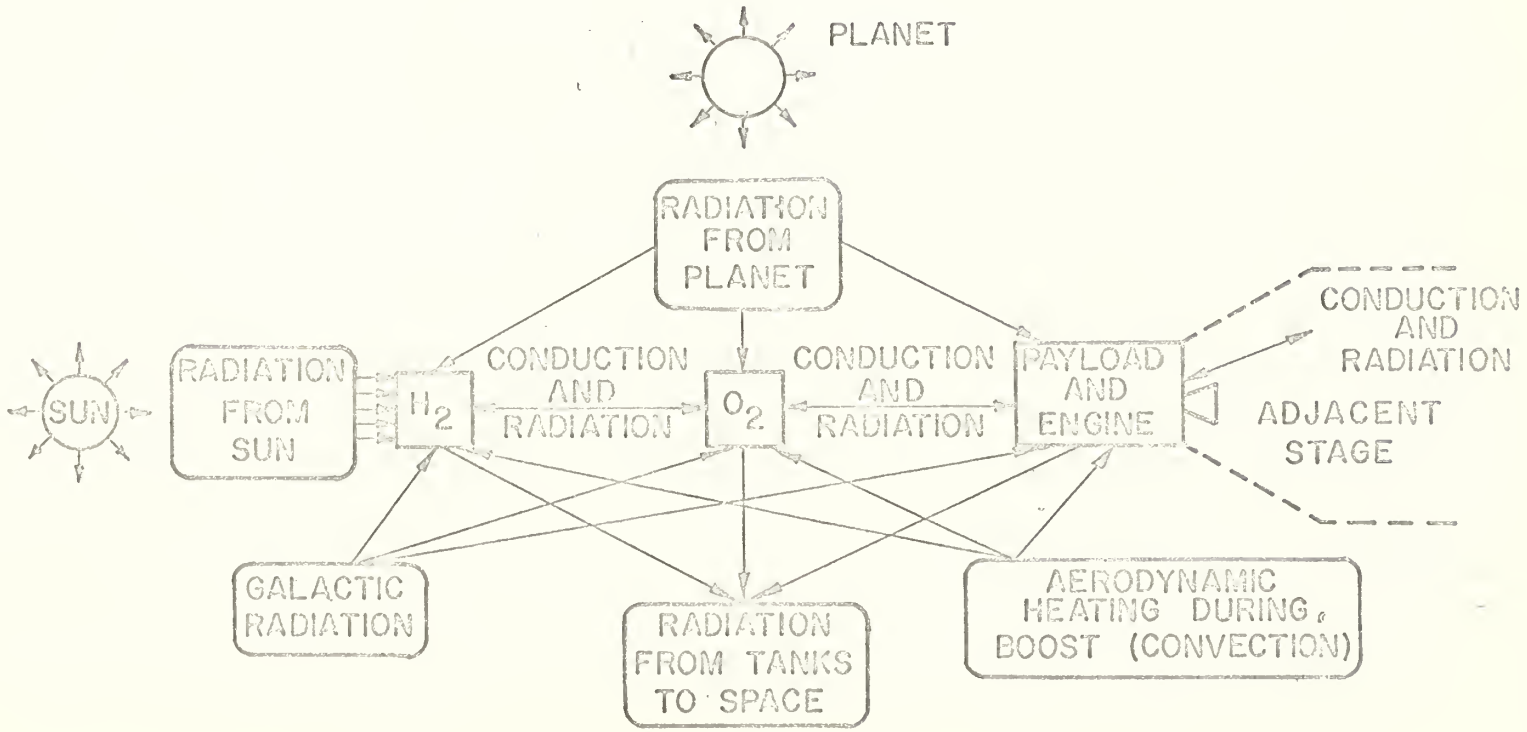
Although cryogenic liquids (such as hydrogen) are among the best propellants currently available for both chemical and nuclear rocket stages, their low temperature characteristics present special problems in the design of the tanks and associated fluid systems. During the course of an interplanetary space mission, heat transfer to these cryogenic liquids from the sun, planets, planet atmospheres, and from other components of the rocket vehicle is inevitable. This heating causes propellant vaporization and consequent loss by venting. Unless these losses are small, the potential advantage of using cryogenic propellants would be negated. Thus, thermal protection of these low temperature liquids from an adverse heating environment is required.

The tank problem is illustrated in more detail in Fig. 15. Here are shown the various sources and sinks for heat flow to and from the propellants in a space vehicle. Excluding for the moment the aerodynamic heating which is encountered during the boost phase of flight through the atmosphere, the heat sources can be generally classified into those resulting from internal or external conditions.

The onboard sources of heat flux are the adjacent components of the vehicle (i.e., any part of the vehicle to which the propellant will be exposed), and nuclear radiation (assuming a reactor is on board for either propulsion or auxiliary purposes). Heating of cryogenic hydrogen due to adjacent components is caused by thermal radiation and by conduction through propellant lines and structural members. The rate of heating by radiation is approximately proportional to the difference between the fourth powers of the absolute temperature of the adjacent component and the propellant. This can become relatively large if liquid hydrogen at 30° - 40° R or liquid oxygen at 140° R is near a relatively high temperature (room temperature or warmer) component. The rate of heat transfer per unit area by conduction is directly proportional to the product of the temperature difference between adjacent components and the thermal conductivity of the conductor, and inversely proportional to the length of the heat path. Heat transfered by conduction is therefore a function of the design features and detailed structural configuration of each specific vehicle. The structural members that separate and

FIGURE 15

SOURCES OF HEAT INPUT TO PROPELLANTS



and support propellant tanks must be so designed as to ensure low rates of heat conduction. This may be done by using low-conductivity laminated stainless-steel supports; radiation is then the main internal heat transfer mechanism.

The external sources of heat are the sun and the planets. Heat is transferred between these sources and the cryogenic storage system by thermal radiation. The largest external heat flux encountered by a vehicle within our solar system is that which originates from the sun. Because of the great distances from the sun, it is generally assumed that the solar heat flux at the planets consists of parallel waves of electromagnetic radiation. Thus, for a unit area that is perpendicular to a radius vector from the sun, this flux outside the atmosphere of a planet is inversely proportional to the square of the distance from the sun and is given by

$$\frac{Q_H}{A} = \sigma \epsilon_s \left(\frac{r_s}{r_{s,p}} \right)^2 \frac{T_s^4}{144}$$

For the earth the solar heat flux is about 430 btu/hr/sqft. It is obvious then that a hydrogen powered space vehicle's orientation with respect to the sun is critical.

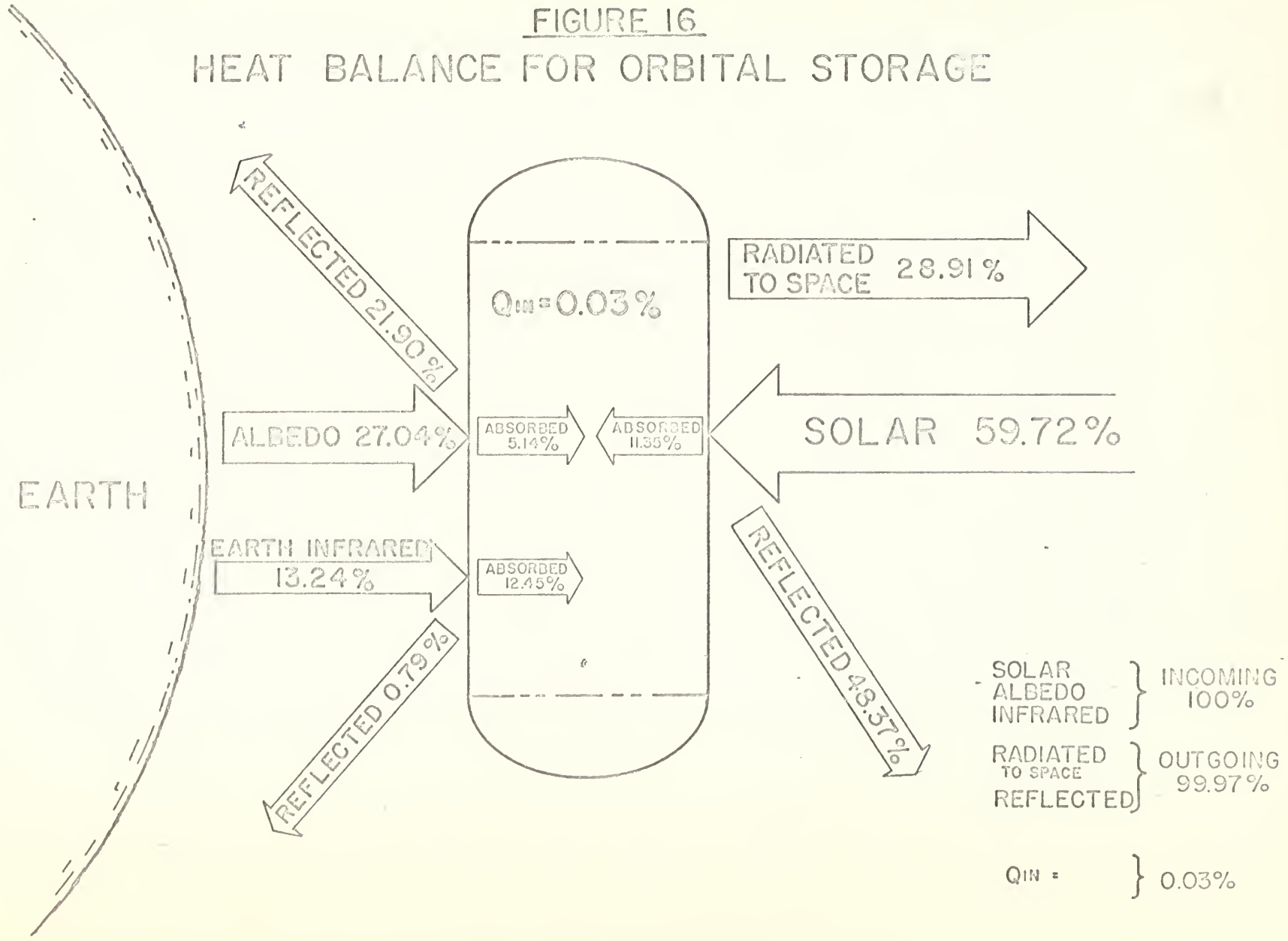
The heat flux that a vehicle receives from a planet results partly from planetary radiation and partly from reflected solar radiation. This heat flux for a body perpendicular to a planet's radii is given by

$$\frac{Q_H}{A} = \frac{1}{144} \left[\sigma \epsilon_p T_p^4 + \sigma \epsilon_s \left(\frac{r_s}{r_{s,p}} \right)^2 T_s^4 \, a_p \, z \right]$$

The factor "a_p", the albedo of the planet, represents the fraction of the incoming solar radiation that is returned to space by scatter and reflections from the planet and its associated atmosphere. The coefficient "z" accounts for the relative position of the body with respect to the planet (z is unity at "noon" and zero at "midnight"). A horizontal surface 100 statute miles above the sunlit earth at noon would receive a planetary heat flux of approximately 234 btu/hr/sqft. Although this heat flux is relatively large, it never exceeds that from solar radiation. The galactic heat flux is extremely small, and when compared to

FIGURE 16

HEAT BALANCE FOR ORBITAL STORAGE



the heat leaks realistically expected in the storage of liquid hydrogen, it becomes insignificant. This flux has a value of approximately 1.8×10^{-5} btu/hr/sqft.

The sketch in Fig. 16 is that of a typical heat balance for the tanks of an earth orbiting vehicle. Such a heat balance assumes that an equilibrium skin temperature has been established, and is obviously dependent upon the size, shape, temperature, orientation, and kind of material (absorptivity and emissivity factors) employed in the vehicle. It does illustrate, however, that a continuous influx of heat can be expected to exist in orbital storage. This heat is absorbed by the vaporation of hydrogen and probably of its cryogenic oxidizer. This orization rate can be expected to reduce rapidly with increased distance from the earth. More detailed investigation show that long term storage of liquid hydrogen in space is feasible and does not present undue difficulties.

Cooling Characteristics:

High energy propellant combinations generally create cooling problems because of the associated higher than normal combustion temperatures. These problems are substantially reduced if one of the propellants, the fuel or oxidizer or both, can be used to regeneratively cool the nozzle, combustion chamber, and injector. The remarkable heat capacity of hydrogen makes regenerative cooling an attractive and practical possibility. Fig. 17 compares the cooling capacities of several propellants on the assumption that the mixture ratio corresponds to maximum specific impulse and that the coolant will be used over reasonable lower and upper limits of temperature. For the oxidizers shown on the figure, the lower shaded portion of the bar is for conditions when no boiling occurs, whereas the upper portion of the bar is for cooling with boiling. The large specific heat and the wide operating temperature range for hydrogen give it a superiority in spite of the generally low fuel to oxidizer ratio of hydrogen propellants.

One method of comparing the cooling capability of various propellant combinations is through a parameter defined as the ratio of the cooling requirement of an engine to the cooling available from the propellant being

FIGURE 17

COOLING CAPACITIES

CHAMBER PRESSURE, 300 PSIA MAXIMUM SPECIFIC IMPULSE

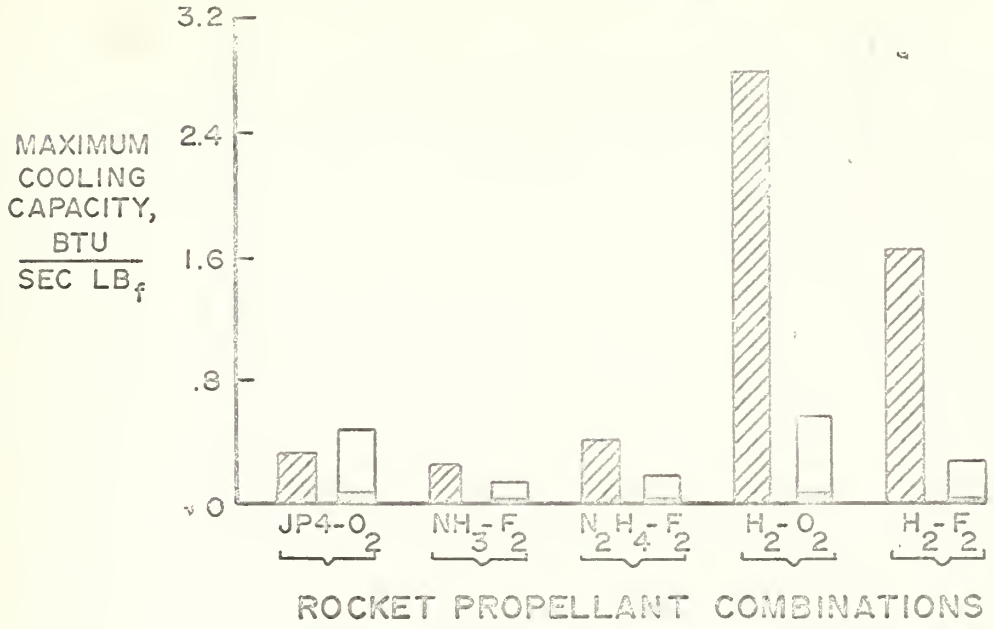
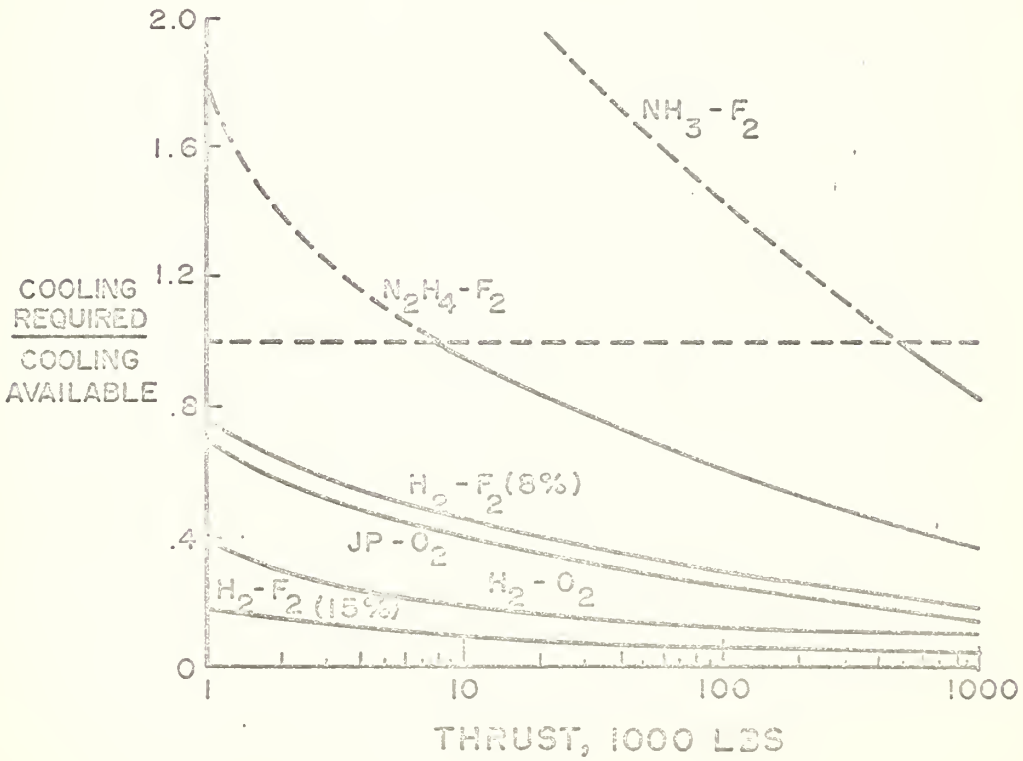


FIGURE 18

COOLING CAPABILITIES

CHAMBER PRESSURE, 300 PSI



used. Theoretically, if the value of this parameter exceeds unity the engine cannot be regeneratively cooled. Fig. 18 shows this cooling capability as a function of engine thrust for several propellant combinations. The superiority of hydrogen for regenerative cooling is evident. Although a H_2-F_2 system (with 8% H_2 by weight, for maximum specific impulse) requires considerable more cooling than the RP-1-LOX system, it has a cooling capability almost equal to that of the RP-1-LOX combination. By increasing the weight flow of the hydrogen in a H_2-F_2 combination from 8 to 15 % the cooling capabilities of the system are greatly increased.

A mathematical analysis in Appendix B demonstrates that hydrogen has high film heat transfer coefficients. Even though the velocity of the hydrogen in the cooling passages is some 20 times higher than that of conventional RP-1-LOX propellants, these high heat transfer coefficients are not associated with excessive pressure drops.

Pumps:

Low weight and high thrust in chemical propelled rocket engines is possible if the propellants have low pressures in the tanks associated with high pressures in the combustion chamber. Thin-walled, light weight tanks can then be designed. The component which satisfies these two extremes of pressure is the turbopump unit. The primary requirement of such a unit is that it produce a high pressure rise with good efficiency and reliability, yet be of light weight construction. Its function is to take the low pressure propellant from the storage tank and deliver it at a high pressure to the combustion chamber while additionally maintaining the pressure needs of metering and control, regenerative cooling, injection and atomization, and fluid line losses. The weight of the turbopumps and associated accessories can be as much as one-half of the entire rocket engine weight; in any case a pound of pump reduces the payload by one pound. Weight of the turbopump varies inversely with the rotational speed (i.e., as speed increases the required diameter of pump and drive turbine decreases). Therefore, increasing rotational speed will lower both pump and drive turbine weights.

The pressure rise is usually described in terms of the head of fluid delivered. This head at pump exit is equivalent to the work input which

depends on theperipheral speed of the rotor, the change in tangential velocity component of the fluid, and the pump efficiency.

$$\text{HEAD DELIVERED} = \Delta H_{is} = \Delta H_{\text{work in}} \times \eta$$

$$\Delta H_{is} = \frac{U_2 V_{u2}}{g} \eta \text{ (ft)} \qquad \text{for } V_{u1} = 0$$

$$\Delta H_{is} = \frac{U_2^2}{g} \left(\frac{V_{u2}}{U_2} \eta \right) \text{ (ft)}$$

$$H = \Delta H_{is} = \frac{U_2^2}{g} \Psi \text{ (ft) Where } \Psi \text{ (press coef)} = \frac{V_{u2}}{U_2} \eta$$

"H" is simply the total head in feet that any specified pump will produce. The pressure increase delivered is dependent upon the density of the particular fluid being pumped.

$$\Delta P = H\rho \qquad (1)$$

A limitation to increasing this head by increasing the peripheral speed of the rotor of a pump occurs because of cavitation. As speed increases, deterioration of pump performance generally takes place because of cavitation before mechanical or stress limitations of the rotor are reached. Cavitation can be described as a flow condition within the pump at which the local pressure reaches or drops below the vapor pressure of the fluid, causing vapor bubbles (boiling) to be formed. As the vapor bubbles reach regions of higher pressure, they collapse and cause structural damage in the form of erosion, as well as the deterioration of pump performance. A conventional parameter indicating the tendency of geometrically similar pumps to cavitate is suction specific speed.

By definition

$$S = \frac{NQ^{\frac{1}{2}}}{H_{sv}^{3/4}} \qquad (2)$$

Specific speed depends upon the difference between the pump inlet pressure and the vapor pressure of the fluid (net positive suction head, H_{sv}), as well as the rotational speed and the volume fluid flow rate. Rocket pumps passing conventional propellants can operate at suction specific speeds approaching 30,000; however, hydrogen pumps, because of the unique physical properties of the fluid merit additional discussion.

It appears, after examining equation (1), that because the density of liquid hydrogen is approximately one-sixteenth that of either liquid oxygen or liquid fluorine, the hydrogen pump will have to produce sixteen times as much head. To do so requires either higher rotational speed, more stages, or a larger diameter (for higher peripheral velocity); the latter two result in more pump weight. If the rotational speed is limited by cavitation of the fluid, an increase in pressure rise will always be associated with an increase in weight. The added pump weight to produce the required head will vary with suction specific speed approximately as shown in Fig. 19. Pump suction pressure is inlet pressure minus fluid vapor pressure.

Expansion of equation (2) by the relationships

$$H_{sv} = \frac{\Delta P_{sv}}{\rho} \quad \text{and} \quad Q = \frac{w}{\rho}$$

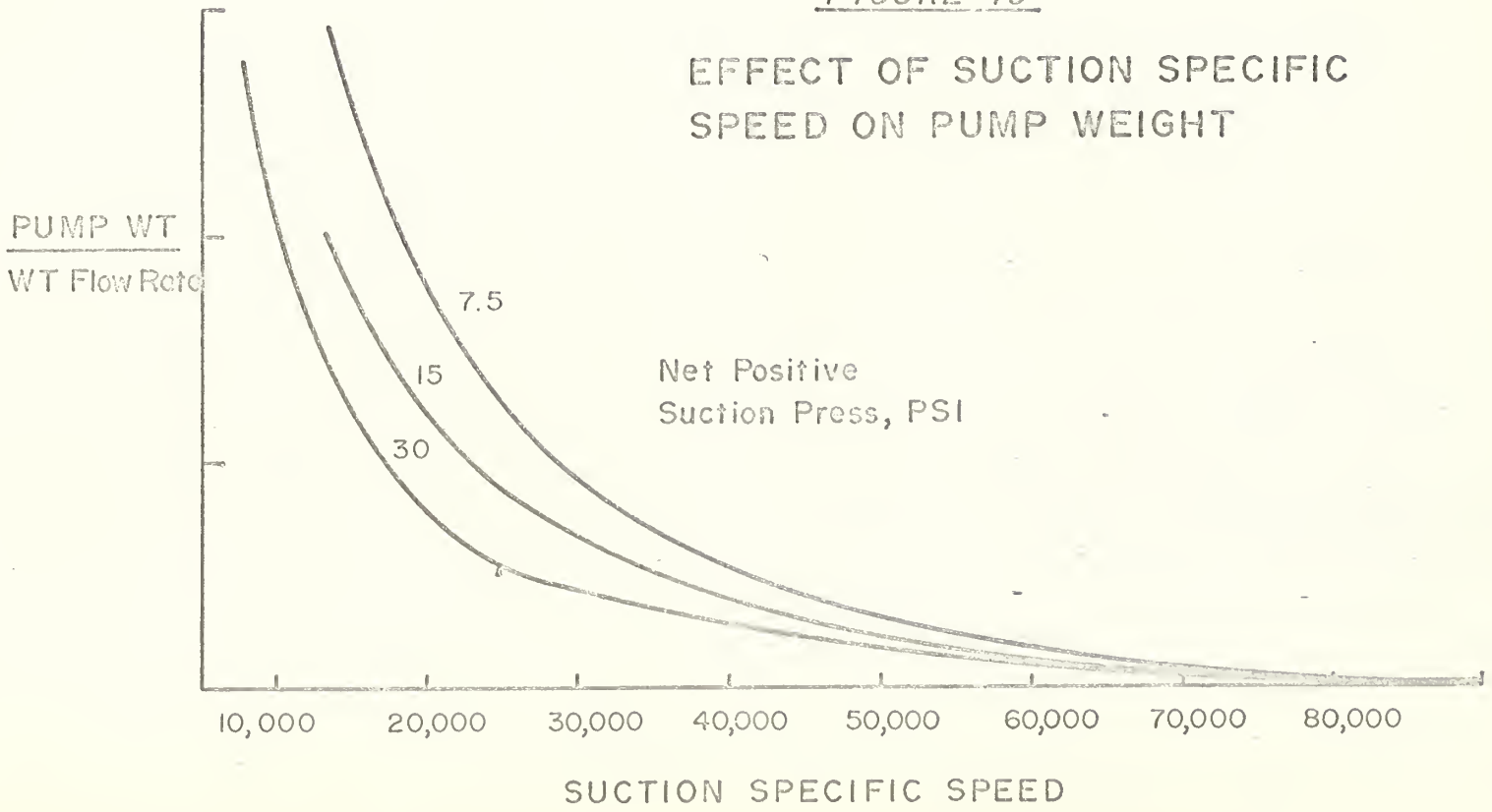
gives

$$S = \frac{N(w)^{\frac{1}{2}} \rho^{\frac{1}{4}}}{P_{sv}^{3/4}}$$

It now appears that the low density of liquid hydrogen will permit some increase in rotational speed and thus in peripheral rotor speed at a given value of suction specific speed and propellant weight flow. It is also possible that the net positive suction pressure be increased by enlarging the inlet area at a given flow rate, thus, allowing a higher RPM at a given S. More important is that liquid hydrogen possesses certain unique properties such as low liquid density, low viscosity, and low boiling temperature which contribute directly to a lessened tendency of this fluid to cavitate. These properties of liquid hydrogen permit operation with little or no increased loss in head because of flow

FIGURE 19

EFFECT OF SUCTION SPECIFIC SPEED ON PUMP WEIGHT



deterioration at suction specific speeds much higher than those at which conventional fluids show large losses. In addition liquid hydrogen does not cavitate with the violent action which is observed in normal fluids. Its tendency to "foam" rather than "boil" lessen its erosive effect and ultimate damage to pump blades. Thus liquid hydrogen pumps are capable of operating at higher specific speeds than pumps for other liquids.

As a consequence, pumps for liquid hydrogen will run at high speeds, usually with stress instead of cavitation limitations. It seems therefore that a hydrogen pump would be larger and heavier than an oxygen or fluorine pump on the basis of propellant weight flow, but not sixteen times as heavy. Fortunately, however, the improved specific impulse and the low percentage of fuel in hydrogen propellant combinations reduce the quantity of hydrogen to be pumped for a given rocket thrust. As a result hydrogen pumps are of about the same size and weight as those used with the more conventional propellants.

Appendix C gives some typical data for comparison on proposed liquid hydrogen and liquid oxygen rocket turbopumps for a 1.2 million pound thrust rocket.

Ignition and Combustion:

The start-up of a rocket engine is an extremely sensitive problem. Very large propellant flow rates can rapidly accumulate unburned explosive mixtures in the combustion chamber. With improper ignition these mixtures can immediately overpressure the chamber, ultimately resulting in structural failure. In a high thrust rocket engine at full flow sufficient propellant can accumulate in a few tenths of a millisecond, and with late or improper ignition, will completely destroy the rocket vehicle.

Hydrogen is hypergolic (spontaneous ignition) with liquid fluorine and most of fluorine's chemical derivatives used as oxidizers. This relieves the problem of providing some additional system for propellant ignition. Under specific conditions of temperature and pressure hydrogen and liquid oxygen are also hypergolic, but not very reliably under all operating conditions. Thus, for a LH2-LOX system the problem exists of providing a reliable, efficient ignition system which is capable of rapidly spreading the flame across the propellants as they enter the chamber.

External pyrotechnics can be used for large booster engines, but this method lacks the re-ignition characteristics that might be required for upper stages. Various torches, that is igniter or spark plugs, can be physically mounted on the injector head and operated several times at command. Another possible ignition source is the injection of a chemical that ignites spontaneously (hypergolic) with either the hydrogen or the oxygen. In addition to furnishing a large amount of ignition energy, the reaction could provide a continuous source of energy during the starting transient and provide restart capability. An ignitor, chemically hypergolic with hydrogen rather than oxygen, is more suitable for this propellant combination because the desired starting sequence requires that the hydrogen enters the combustion chamber before the oxygen to assure regenerative cooling in the thrust chamber at the start of combustion.

Fig. 20 illustrates the transient conditions during the ingition of a liquid hydrogen-liquid oxygen engine by injecting fluorine into the chamber ahead of the oxygen. In this case the fluorine may be either liquid or gaseous; however, the gaseous fluorine is somewhat easier to handle in small quantities. This figure shows a trace of the chamber and feed pressures as they vary with time. Hydrogen flow is started at zero time and is brought up to full flow to insure a cooled chamber. Approximately 1.5 sec. later fluorine is admitted, followed by oxygen which is brought up to full flow. Because the fluorine flow rate is only about 1% that of the oxygen, no noticeable rise in chamber pressure takes place until the oxygen flow is fully established. Once the chamber pressure rises to full value the fluorine flow is stopped.

The ease with which hydrogen can vaporize, in fact hydrogen will probably be a vapor when it enters the combustion chamber, as do oxygen and fluorine, is one of the reasons why higher combustion efficiencies can be achieved with these propellant combinations than with most others. Fig. 21 shows experimental data for a 200 lb thrust rocket engine using several fuels with liquid oxygen and liquid fluorine. Hydrogen produced the highest combustion efficiency at any chamber length with either oxidizer. The behavior of hydrazine with oxygen was shown by other tests to be unusual and possibly attributable to thermal decomposition.

FIGURE 20

TRANSIENT START DATA WITH FLUORINE IGNITION

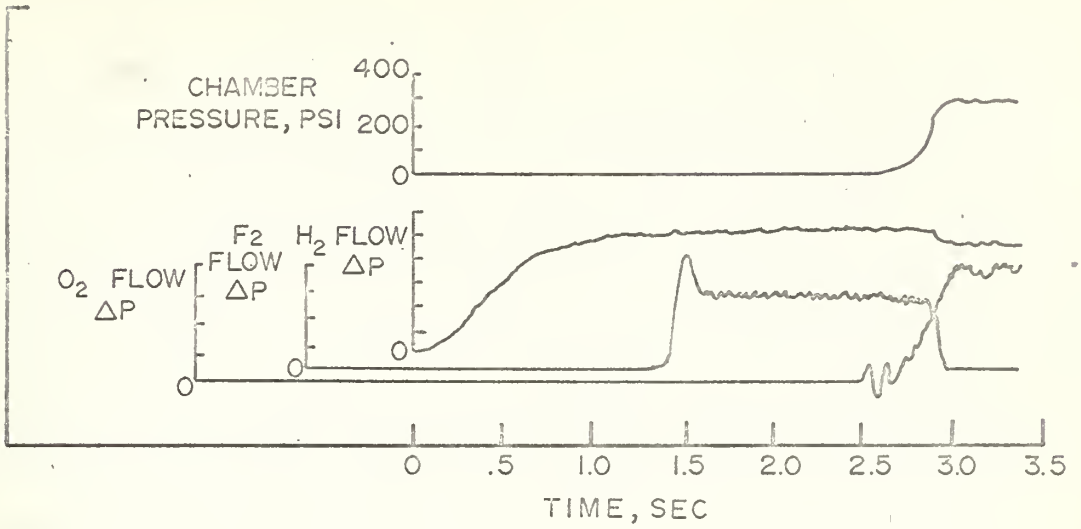
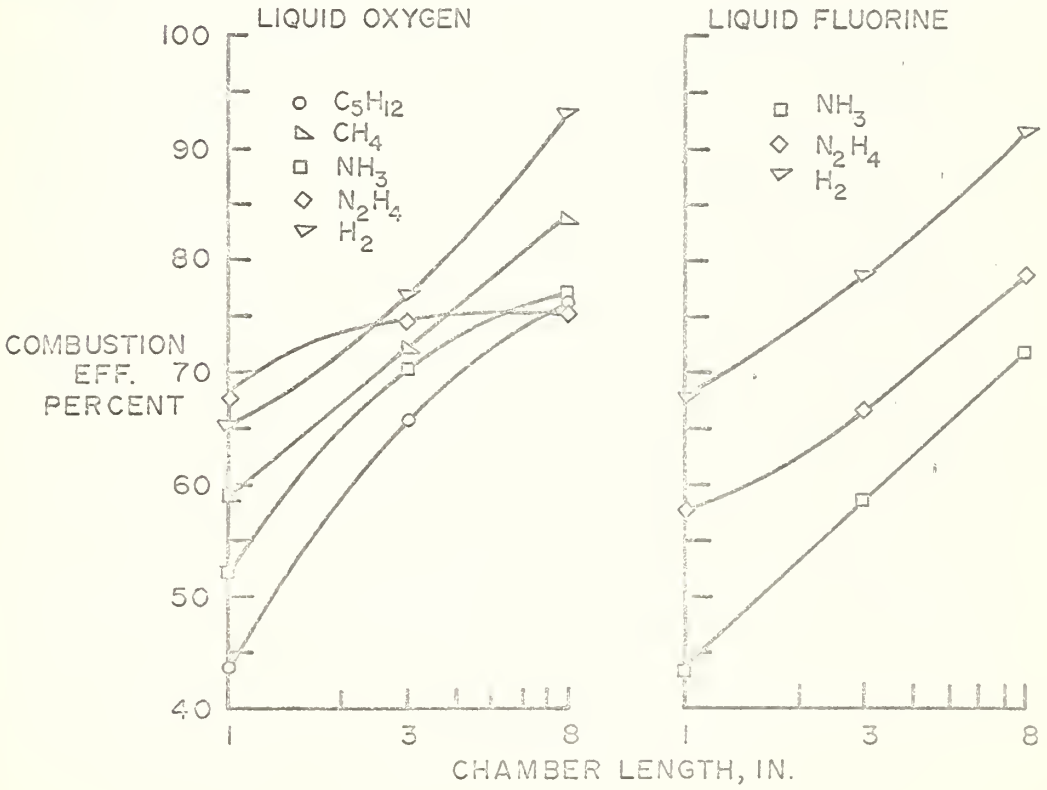


FIGURE 21

COMBUSTION OF O_2-F_2 WITH VARIOUS FUELS



Material Compatibility:

At the low temperatures of liquid hydrogen there are effectively no corrosion problems. Low temperature embrittlement of metals is a difficulty, but materials such as stainless steel, monel, nickel, etc. are available presently for use. There are, however, several anticipated problems when using hydrogen in combination with liquid fluorine. There are no known materials that can withstand the corrosive action of fluorine. Even teflon, when used as a seal, will disintegrate under a steady stream of liquid fluorine. Table III lists materials compatible with liquid hydrogen and liquid oxygen respectively for various systems, i.e. tanks, valves, etc.

Handling and Safety:

The use of liquid hydrogen as a low temperature fluid and as a high energy fuel introduces several hazards not ordinarily associated with other cryogenic fluids. These hazards are attributable to the many unique properties of this material, either as a gas or liquid.

The chief hazards associated with the use of hydrogen in unconfined (vented) spaces are (1), those attributed to the formation of shock sensitive (liquid hydrogen-solid oxygen or air) mixtures and (2), fire. In the absence of an oxidizer, liquid hydrogen is quite stable; therefore, the first-mentioned hazard can be eliminated by excluding oxygen from all systems with which liquid hydrogen comes in contact. The most obvious solution is to flush such systems with an inert gas, such as nitrogen or helium. Since trace quantities of oxygen are difficult to remove from the hydrogen gas to be liquified, solid oxygen deposits may be built up over a period of time in storage containers from which liquid hydrogen is withdrawn. Periodical purging and cleaning of these containers might well be necessary. Because liquid hydrogen cannot be stored in a sealed container, the second hazard (fire) can only be minimized by the judicious placement of vent stacks, combustibles, equipment, and buildings. There is a wide range of mixture compositions over which flammable hydrogen-air mixtures can be formed. Associated with these mixtures are very high burning velocities; therefore, flame stacks should not be used to dispose of the excess hydrogen. Quantity-distance tables are available which recommend the safe minimum distance

TABLE III

MATERIAL COMPATIBILITY WITH LIQUID OXYGEN AND LIQUID HYDROGEN

	TANKS		LINES		VALVES		FITTINGS		O-RINGS		GASKETS	
MATERIAL	O_2	H_2	O_2	H_2	O_2	H_2	O_2	H_2	O_2	H_2	O_2	H_2
Titanium		X		X		X		X		X		
Austenitic Stainless Steel	X	X	X	X	X	X	X	X	X	X		
Aluminum Alloys	X	X	X	X	X	X	X	X	X	X		
Carbon Steel	Low Temperature Embrittlement											
High Strength Alloy Steel	Marginal Low Temperature Ductility											
Fiberglass	X (Filament Wound)	X										
Polyethylene	Embrittlement at LO_2 Temperature											
Teflon								(Critical Design)				
Natural and Syntehtic Rubber								Too Brittle				
Asbestos											X (Non-Lubricated)	
Brass					X	X	X	X				

for inhabited buildings and additional storage tanks based upon the quantity of liquid hydrogen stored in any one container. These minimum distances for buildings range from 100 to 350 ft. for quantities of 200 to 10,000 lbs. respectively, and from 50 to 300 ft. between storage tanks for quantities of from 2,000 to 100,000 lbs.

Transportation of liquid hydrogen may be effected by highway semi-trailer or air-transport of such a trailer or portable dewar. Both methods have been proved safe providing the proper precautions are taken.

For the safe and efficient handling of liquid hydrogen both the cryogenic and chemical properties must be considered, and the associated problems must be met prudently. If this is done, the use of liquid hydrogen can be realized to its full potential as a rocket fuel.

Table IV summarizes the storage and handling safety precautions.

Cost and Availability:

The use of liquid hydrogen as a rocket fuel is predicated by its availability and cost. Liquid hydrogen is not inexpensive when compared to RP-1, common kerosene, but neither is it expensive when priced against some of the conventional storable propellants such as hydrazine (N_2H_4). Table V lists several rocket propellants and compares their present availability and cost against that projected in the near future. Note that with the increased use and production of liquid hydrogen the costs are expected to decrease markedly.

TABLE IV

STORAGE & HANDLING SAFETY PRECAUTIONS

OF LIQUID HYDROGEN

STORING:

(1) On the whole, liquid hydrogen may be regarded as hazardous as a highly volatile gasoline. The primary precaution is the prevention of air leakage into containers of liquid hydrogen. The air would freeze, providing a combustible combination if a flame or spark were present. A possible source of such a spark is the breaking of a crystal of solid air or oxygen.

(2) Proper precautions should be taken to provide for adequate venting of the hydrogen vapor which prevents tank pressure build-up.

(3) The three steps which provide adequate protection against explosion are:

(a) Careful pressure and leak testing of all lines and equipment.

(b) Adequate ventilation to exhaust vent gas.

(c) Elimination of the likelihood of a flame or spark occuring in the hydrogen area.

(4) Safety requires a combustible alarm system to analyze gas samples from critical locations of a ground storage facility.

(5) To minimize spark or flame occurance, the following measures are required:

(a) All electrical wires and machinery shall be explosion-proof.

(b) All tools shall be spark-resistant beryllium-copper.

(c) All belt driven machinery shall be equipped with spark-resistant belts.

TRANSPORTING:

(1) Highway semi-trailer 1,500 to 6,000 gal.

(2) Air transport of trailer or portable dewar.

Transfer of liquid hydrogen must be accomplished by insulated lines to prevent excessive loss. Evaporated hydrogen is used to produce the required pressure head. The most critical item is attaining leak-tight connections for safe, efficient transfer.

TABLE V

PROPELLANT LOGISTICS

Propellant	Current		Projected	
	Quantities	Cost	Quantities	Cost
N_2O_4	1 ton lots	7.5 cents/lb	tank car lots	6.5 cents/lb
CLF_3	unknown	$2.50 to 3.00/lb	100 ton/year	$0.50/lb
N_2H_4	drum lots	$3.00/lb	tank car lots	$0.50/lb
UDMH	tank lots	$2.00/lb	tank car lots	$0.50/lb
Liquid Oxygen	Very Large Quantities Currently Available	2.3 to 5.6 cents/lb	Continue Very Large	No Change Expected
Liquid Oxygen	75,000 lbs/ day production	75 cents/lb	in excess of 75,000 lbs/day	$0.25 to $0.50/lb

5. CONCLUSION

Although the work done in connection with the present thesis had to be confined primarily to a survey of the literature, its undertaking was considered a valuable endeavor.

In surveying the subject literature and compiling the appropriate information much was learned of the capabilities, problems, and condition of the present state of the art of liquid hydrogen rocket boosters. In addition much knowledge was gained in a more general manner during sub-investigations which were required in order to fully understand some of the available reference material. The short time available for its completion minimized the depth and detail to which this thesis could be taken.

It is concluded that liquid hydrogen offers the most attractive propellant fuel presently available for space booster systems. Although several problems not previously encountered with other propellants present themselves in these systems, no new "breakthroughs" are required and no problems appear unsolvable. Increased flight testings of hydrogen fuel in actual booster systems will ultimately lead to the realization of the full potential of this high energy propellant. The faith of the National Aeronautics and Space Administration in liquid hydrogen used as the major propellant component of its Saturn V lunar payload booster is shared, and the justification of this faith is hopefully anticipated.

BIBLIOGRAPHY

1. Sutton, G. P; Rocket Propulsion Elements, John Wiley & Sons, New York, 1956

2. Fitzgerald, F; "High Energy Chemical Propellants", Spaceflight, Nov. 1961: p199-203, 6

3. Coar, R. J., King, C. H; "Hydrogen for the Space Age", Astronautics, March 1960: p26-27, 76-78

4. Carter, J.M; "Isp: How High for Chemical Fuels?", Astronautics, Sept. 1958: p26-27, 74-76

5. Olson, W.T; "Problems of High-Energy Propellants for Rockets", Chemical Engineering Progress Symposium, v57 n33 1961: p28-37

6. Christian, J. C., Chafey, J. E., Harlich, A., Watson, J. F., Witzell, W. E; "Structural Alloys for Cryogenic Service", Metal Progress, March 1963: p101-104, 138, 142

7. Roder, H. M., Diller, D.E., Weber, L. A., Goodwin, R. D; "The Orthobaric Densities of Parahydrogen, Derived Heats of Vaporization, and Critical Constants", Cryogenics, March 1963: p16-222

8. Stambler, I; "Centaur", Space/Aeronautics, Oct. 1963: p70-75

9. Fox, R. H; "Nuclear Propulsion", Astronautics and Aerospace Engineering, Nov. 1963: p104-106

10. Zwicky, F; "Propellants for Tomorrow's Rockets", Astronautics, Aug. 1957: p45-49

11. Mitchell, E; "Hydrogen-LOX Development Exceeds Forecasts", Missiles and Rockets, July 1960: p71-84

12. Hendel, F. J; "Advanced Rocket Propulsion", Chemical Engineering, April 1961: p131-148

13. Lange, Dr. O. H; "Trends in the Development of U.S. Space Vehicles", Interavia, No. 12, 1961: p1636-1650

14. "NASA'S Ten Year Programme"; Interavia, No. 10, 1962: p1278-1283

15. "Preliminary Design of Spacecraft Liquid Propulsion Systems"; Beech Aircraft Corporation, Report 9227, Sept 1960:

16. "Large Rockets"; IAS meeting proceedings, Oct 1962:

17. Dobbins, T. O., 1/Lt USAF; "Thermodynamics of Rocket Propulsion & Theoretical Evaluation of Some Prototype Propellant Combinations", Wright Air Development Center, Wright-Patterson Air Force Base, Dec 1959:

18. "Experimental Heat Transfer and Pressure Drop of Liquid Hydrogen Flowing Through a Heated Tube"; NASA Technical Note D-765, May 1961:

19. "Orbital Storage of Liquid Hydrogen'; NASA Technical Note D-559, Aug 1961:

20. "Ground Facility Requirements for Subcooling Liquid Hydrogen"; NASA Technical Note D-1276, July 1962:

21. "Analysis of Thermal - Protection Systems for Space Vehicle Cryogenic - Propellant Tanks"; NASA Technical Report R-130, 1962:

22. "Cavitation Performance of an 83° Helical Inducer Operated in Liquid Hydrogen"; NASA Technical Memo x-419, Mar 1961: conf

23. "Performance Characteristics of Hydrogen-Fluorine Rocket Engine at Various Chamber Pressures and Nozzle Area Ratios"; Bulletin of the Second Meeting of the Joint Army-Navy-Air Force Liquid Propellant Group, Nov 1960: conf

24. "Propulsion Systems for Our Future Spacecraft"; Bulletin of the Fourth Army-Navy-Air Force - NASA - ARPA Liquid Propulsion Symposium, Nov 1962: conf

25. "Comparisons of Nuclear, Electrical and Liquid Propellant Propulsion Systems for Space Applications"; Bulletin of the Second Meeting of the Joint Army-Navy-Air Force Liquid Propellant Group, Nov 1960: conf

26. "Performance Characteristics of Hydrogen-Fluorine Rocket Engines at Various Chamber Pressures and Nozzle Area Ratios"; Bulletin of the Second Meeting of the Joint Army-Navy-Air Force Liquid Propellant Group, Nov 1960: conf

27. "Explosion Hazards of Liquid Hydrogen"; Bulletin of the Second Meeting of the Joint Army-Navy-Air Force Liquid Propellant Group, Nov 1960: conf

28. "Ignition of Hydrogen-Oxygen Rocket Engines with Fluorine"; NASA Technical Memo X-101, Dec 1959: conf

APPENDIX A, B, C, D

APPENDIX A

Chemical Energy Release from Combustion

Rockets utilize the heat liberated (heat of reaction) in the combustion of chemical propellants as a source of energy. The quantity of energy available from a given propellant combination is determined by the chemical nature of the oxidizer and fuel molecules as well as by the nature of the reaction gas products. The propellant molecules should be as weakly valence bonded as is compatible with stability and good physical properties. Compounds whose formation is characterized by an absorption of energy will generally liberate this additional energy on combustion, thus providing increased performance.

Definitions:

Heat of Reaction (ΔH_R) - the change in enthalpy which occurs when products are formed from reactants at standard conditions, namely at a constant reference temperature and pressure.

Heat of Formation (ΔHf) - the change in enthalpy which results when a compound is formed at standard conditions from its elements isothermally at constant pressure.

Table A-I lists heats of formations at given reference temperatures.

$$\Delta H_R = \left[\Sigma\, n\, \Delta\, H_f\right]_{\text{products}} - \left[\Sigma\, n\, \Delta\, H_f\right]_{\text{reactants}}$$

The above equation describes the chemical energy release for a typical rocket combustion process. Not included are the energy requirements to heat the reactants from their initial temperature to the reference temperature at which ΔH_R is calculated, and from that temperature to the combustion flame temperature. The desired results of such a process are that ΔH_R have a large negative (exothermic) value. The heat of formation of the products is a large negative number while that of the reactants generally has a positive or small negative value. It is obvious that the latter condition of the reactants increases the available energy release.

TABLE A-1

HEATS OF FORMATION

Compound*	Temperature- °F	ΔH_f - BTU/lb-mole
C	77	+309,060
CO	77	-47,550
CO_2	77	-169,300
H	32	+92,910
H_2O	77	-104,000
N	32	+153,200
NH_3	77	-19,870
N_2H_4(liquid)	77	+21,690
NO	32	+38,660
NO_2	77	+14,560
O	32	+105,400
OH	32	+18,000
N_2, H_2, O_2, Cl_2, He	77	0
HNO_3 (liquid, 100% concentration)	77	-74,530
C_2H_5OH (100% concentration)	77	-101,232
C_8H_{18}	77	-96,430
H_2O_2 (liquid, 90% concentration)	77	-81,020
CH_2	77	+134,900

*gas unless otherwise denoted

The following calculations show the high comparative heat content available from hydrogen. These reactions do not take account of the dissociation of the combustions products. Since this dissociation is a function of the pressure and combustion flame temperature they are omitted for simplification. This introduces no error when using the results on a comparative basis only.

Combustion of hydrogen:

$$H_2 + \tfrac{1}{2} O_2 \longrightarrow H_2O \text{ (Stoichiometric)}$$

$$\Delta H_R = \Delta H_{f_{H_2O}} - \left[\Delta H_{f_{H_2}}^{\circ} + \tfrac{1}{2} \Delta H_{f\ O_2}^{\circ} \right]$$

$$\Delta H_R = \left(-104{,}000 \text{ BTU/mole } H_2O\right) - 0 \text{ (for 1 mole } H_2 \text{ burned)}$$

$$\Delta H_R = -52{,}000 \text{ BTU/lb } H_2 \text{ Burned}$$

Combustion of energetic hydrocarbon:

$$CH_2 + 3/2\ O_2 \longrightarrow CO_2 + H_2O \text{ (Stoichiometric)}$$

$$\Delta H_R = \left[\Delta H_{f_{CO_2}} + \Delta H_{f_{H_2O}} \right] - \left[\Delta H_{f_{CH_2}} + 3/2\ \Delta H_{f_{O_2}}^{\circ} \right]$$

$$\Delta H_R = \left[-169{,}300 \frac{\text{BTU x 1 mole } CO_2}{\text{mole } CO_2} - 104{,}000 \frac{\text{BTU x 1 mole } H_2O}{\text{mole } H_2O} \right]$$

$$- \left[+134{,}900 \frac{\text{BTU x 1 mole } CH_2}{\text{mole } CH_2} + 0 \right]$$

$$\Delta H_R = -408{,}200 \text{ BTU/mole } CH_2 \text{ Burned}$$

$$\Delta H_R = -29{,}200 \text{ BTU/lb } CH_2 \text{ Burned}$$

In actuallity, because of high temperatures, dissociation of the gaseous exhaust products does occur until an equilibrium condition is reached at the flame temperature and chamber pressure. If this equilibrium is maintained while these exhaust gases are expanded through the nozzle a condition exists which is usually called frozen equilibrium. The I_{sp} calculated for these conditions is termed the frozen specific impulse. If, however, because of changing temperatures and pressures as flow passes through the nozzle, the equilibrium shifts, additional energy is liberated as shown below. The I_{sp} calculated for these conditions is called the shifting or theoretical specific impulse.

Possible exhuast gas products from hydrogen-oxygen reaction at some equilibrium condition are given by the following decomposition:

$$a\ H_2 + b\ O_2 \longrightarrow c\ H_2O + d\ H_2 + e\ O_2 + f\ H + g\ O$$

As pressure and temperature decrease during flow through the nozzle:

$H \longrightarrow \frac{1}{2} H_2$ exothermic

$O \longrightarrow \frac{1}{2} O_2$ exothermic

$H_2 + \frac{1}{2} O_2 \longrightarrow H_2O$ exothermic

If complete re-association takes place, the maximum energy potential is released which leads to calculation of the maximum theoretical impulse obtainable. In reality neither frozen nor maximum theoretical equilibrium conditions exist in the flow through a nozzle. The speed at which the exhaust gases travel, and thus time, is the deciding factor that determines the amount of additional energy release in the nozzle. For complete re-combination theoretically an infinitely long nozzle would be required. Therefore, it is obvious that the maximum performance available from any chemical rocket lies between its frozen and theoretical equilibrium values of specific impulse.

APPENDIX B

Film Heat Transfer Coefficient

Fig. B-1 assumes a fluid flowing through a tube, upon the wall of which a shear stress, τ, exists. A random particle, of elemental mass and velocity, V, and temperature, T, within the fluid comes to rest within the boundary layer.

$$dmV - 0 = dF\ dt$$

or

$$\frac{dm}{dt} = \frac{dF}{V} = \frac{\tau\ dA}{V}$$

where dA is the circumferential area of the pipe

The shearing force, τ dA, produces a pressure drop dp, thus

$$\tau\ dA = \frac{\pi}{4} d^2\ dp = \tau\ \pi\ d\ dl$$

and

$$\tau = \frac{d}{4}\ \frac{dp}{dl} \qquad (1)$$

The pressure drop per unit length of a fluid flowing in a tube with friction is given in reference (1) as

$$\frac{dp}{dl} = \frac{f}{d}\ \frac{\rho}{2}\ V^2 \qquad (2)$$

where f is the friction factor for a smooth pipe which, for a turbulent flow is a function of $1/R_e^m$.

Heat absorbed by the coolant in Fig. B-1 is

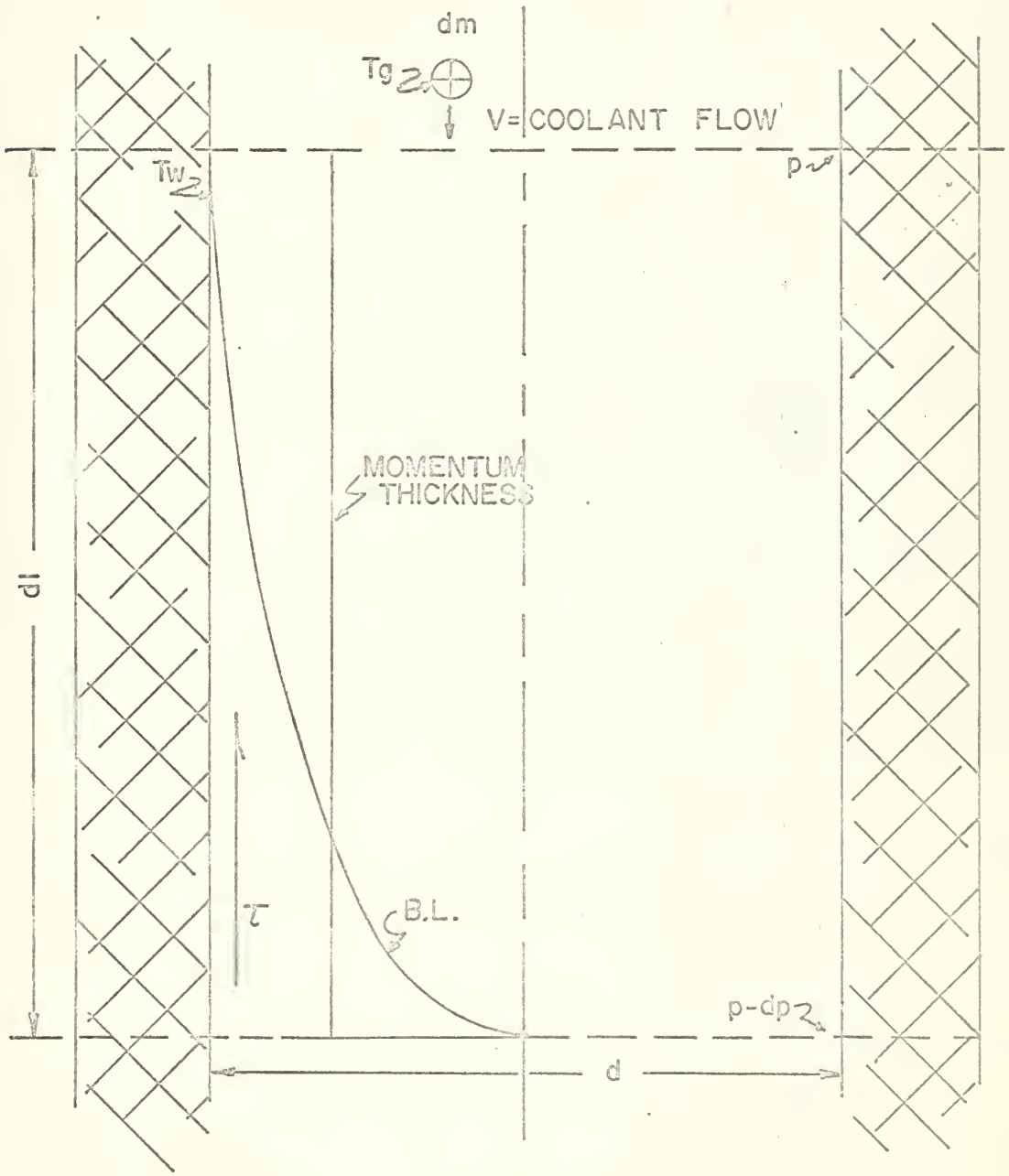
$$\frac{dq}{dt} = \frac{dm}{dt}\ c_p\ (T_w - T_g)$$

$$= dA\ h\ (T_w - T_g)$$

where h is the film heat transfer coefficient in BTU/sec-in^2-$^\circ$F

FIGURE B-1

FLUID THROUGH
A ROUND PIPE



then

$$h = \frac{dm}{dt} \frac{c_p}{dA}$$

$$h = \frac{\tau \, dA}{V} \frac{c_p}{dA}$$

$$h = \frac{\tau \, c_p}{V}$$

From equations (1) and (2)

$$\tau = \frac{\rho}{8} V^2 f$$

therefore

$$h = \frac{c_p}{V} \frac{\rho}{8} V^2 f$$

$$h = \frac{c_p \rho}{8} V f \qquad (3)$$

Equations (2) and (3) are general expressions for the pressure drop per unit length, and film heat transfer coefficient, respectively. They are dependent upon the density, velocity, and friction factor for specific fluids. In addition, the heat transfer coefficient is directly proportional to the specific heat. The friction factor in turn is a function of the Reynolds number, thus depends upon fluid velocity, denisty, and viscosity.

Assuming the same mass flow rate per unit area, ρV, the ratio of the Reynolds number of liquid hydrogen to that of any other coolant fluid simply becomes the inverse ratio of the respective viscosities.

$$\frac{R_e - |H_2|}{R_e - |other|} = \frac{(\rho \, v/\mu) |H_2|}{(\rho \, v/\mu)_{|other|}} = \frac{\mu - |other|}{\mu - |H_2|}$$

From the graph in Fig. B-2 of Reynolds number versus friction factor, assuming an R_e of 10^6 for hydrogen, the ratio of the friction factor for three fluids were calculated and tabulated in Table B-I.

TABLE B-I

COMPARISON OF HYDROGEN AS A COOLANT

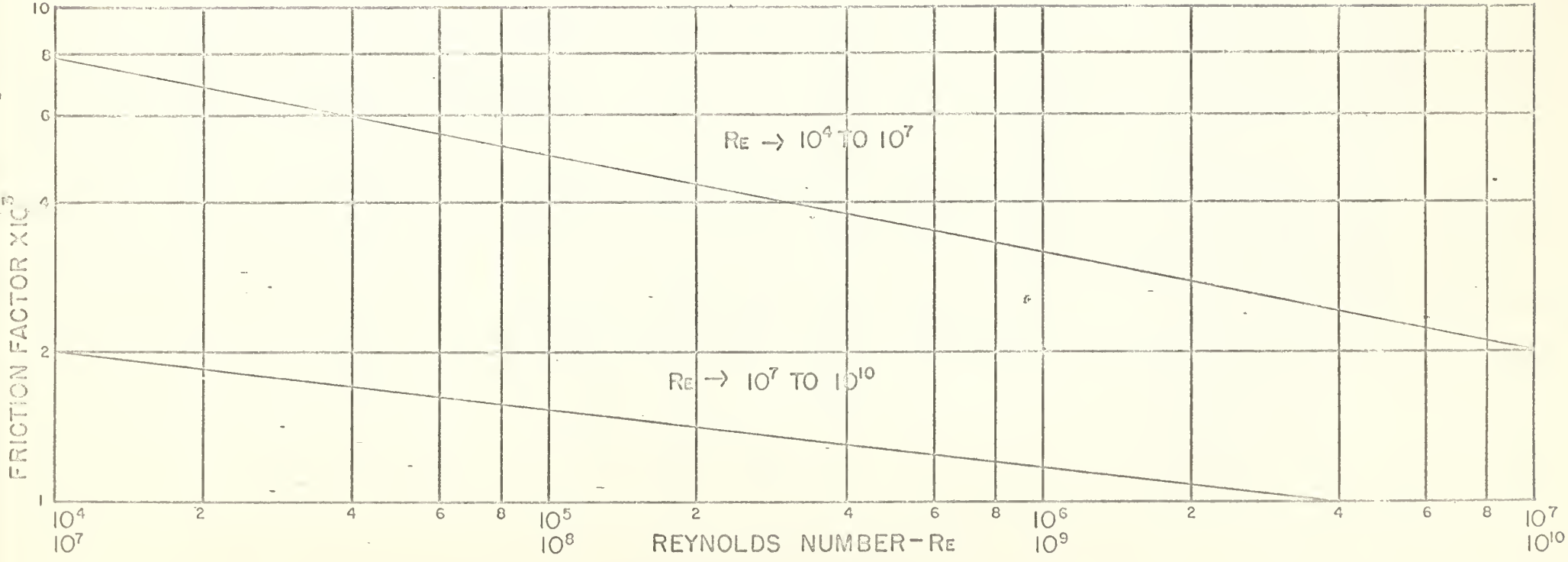
Coolant	μ/μ_{H_2}	f_{H_2}/f	C_{PH_2}/C_P	h_{H_2}/h
H_2	1	1	1	1
O_2	$.19 \times 10^2$	.57	6.6	3.8
N_2H_4	$.71 \times 10^2$	.44	3.2	1.4

Because of its small friction factor and the very low denisty it is possible to resort to higher velocities in the cooling channel for hydrogen, than for other coolants, without an increase in pressure losses.

The film heat transfer coefficient of hydrogen, from equation (3), is also tabulated in Table B-I. Table B-I uses equal mass flow rates as a basis for the comparison of the properties of different coolants. Thus the relative merits of liquid hydrogen are easily recognized. The comparisons of film heat transfer coefficients are very general, and it must be remembered that they depend to a large extent upon the actual mass flow rate of the propellant.

FIGURE B-2

FRICTION FACTOR VS REYNOLDS NUMBER
IN
SMOOTH ROUND PIPE



APPENDIX C

Liquid Hydrogen Turbo-pump Performance

Table C-I presents some important performance parameters of liquid hydrogen and liquid oxygen turbo-pumps for a 1.2 million pound thrust engine. The comparison of these figures shows the requirements and capabilities involved in pumping liquid hydrogen and liquid oxygen.

TABLE C-I

TURBO-PUMP SPECIFICATIONS

		H_2	O_2
TEMP	$^\circ F$	-421	-294
DENSITY	lb/ft^3	4.4	71
SHAFT SPEED	rpm	13,800	3380
TOTAL PUMP DISCHARGE PRESS.	psia	1535	1355
TOTAL INLET PRESS.	psia	29	37
TOTAL HEAD RISE	ft	49,200	2670
WEIGHT FLOW	lb/sec	550	2650
VOLUME FLOW	gpm	56,100	16,700
EFFICIENCY	%	72	67
SHAFT HP	H.P.	68,330	19,150
NET POSITIVE SUCTION HEAD	ft	265	40.5
SUCTION SPECIFIC SPEED	$\frac{rpm\text{-}gpm^{\frac{1}{2}}}{ft^{3/4}}$	50,000	27,300

Fig. C-1 and C-2 show that the reduction in head with increasing suction specific speed is less in a liquid hydrogen pump than in a liquid oxygen pump. Hence, a liquid hydrogen pump can operate at higher peripheral speeds than a liquid oxygen pump. These figures indicate clearly that a liquid hydrogen pump is less susceptible to cavitation than a liquid oxygen pump.

FIGURE C-1

ESTIMATED FUEL PUMP HEAD DROP/NPSH RATIO
V S
SUCTION SPECIFIC SPEED

LIQ H_2 FUEL

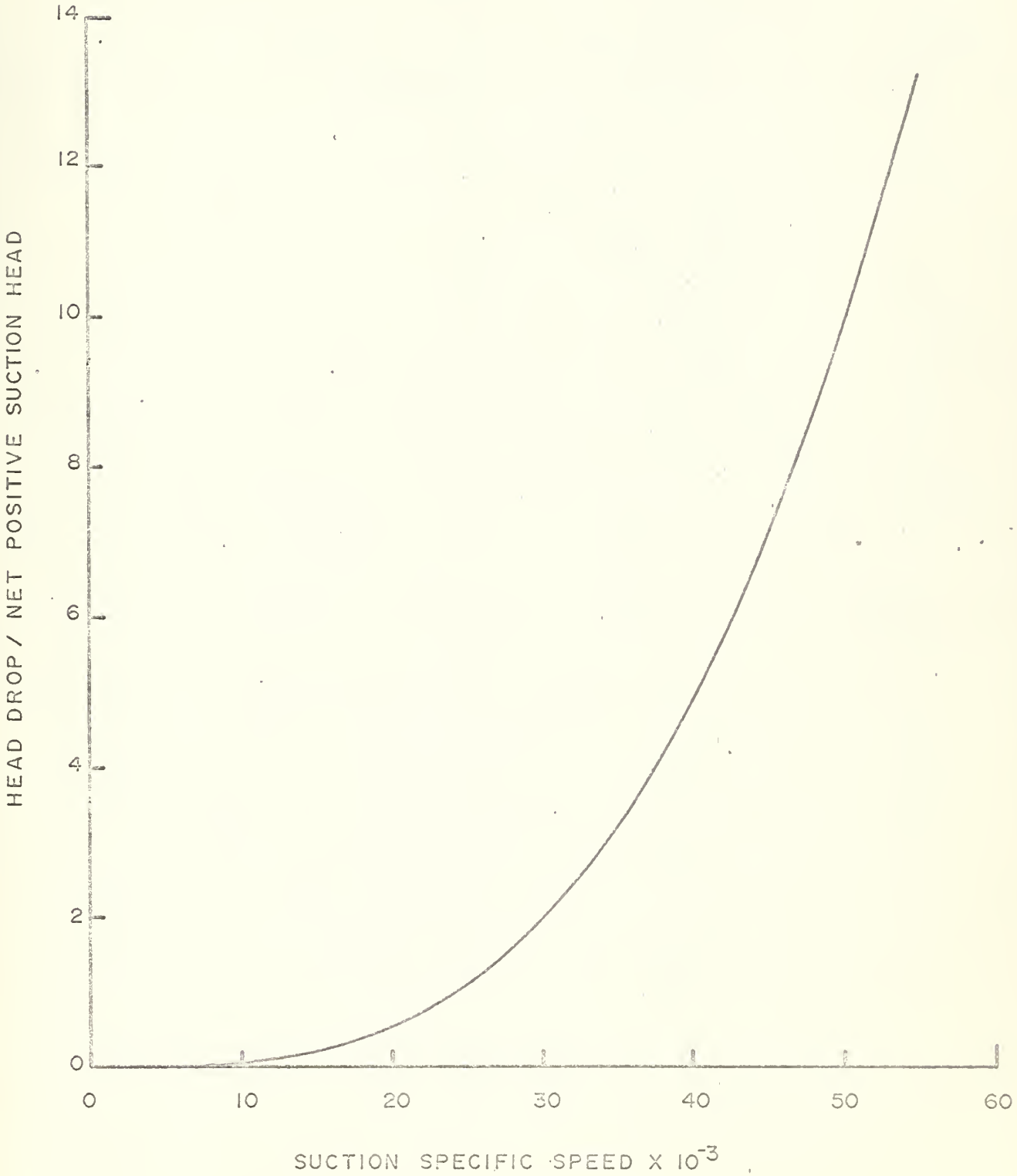
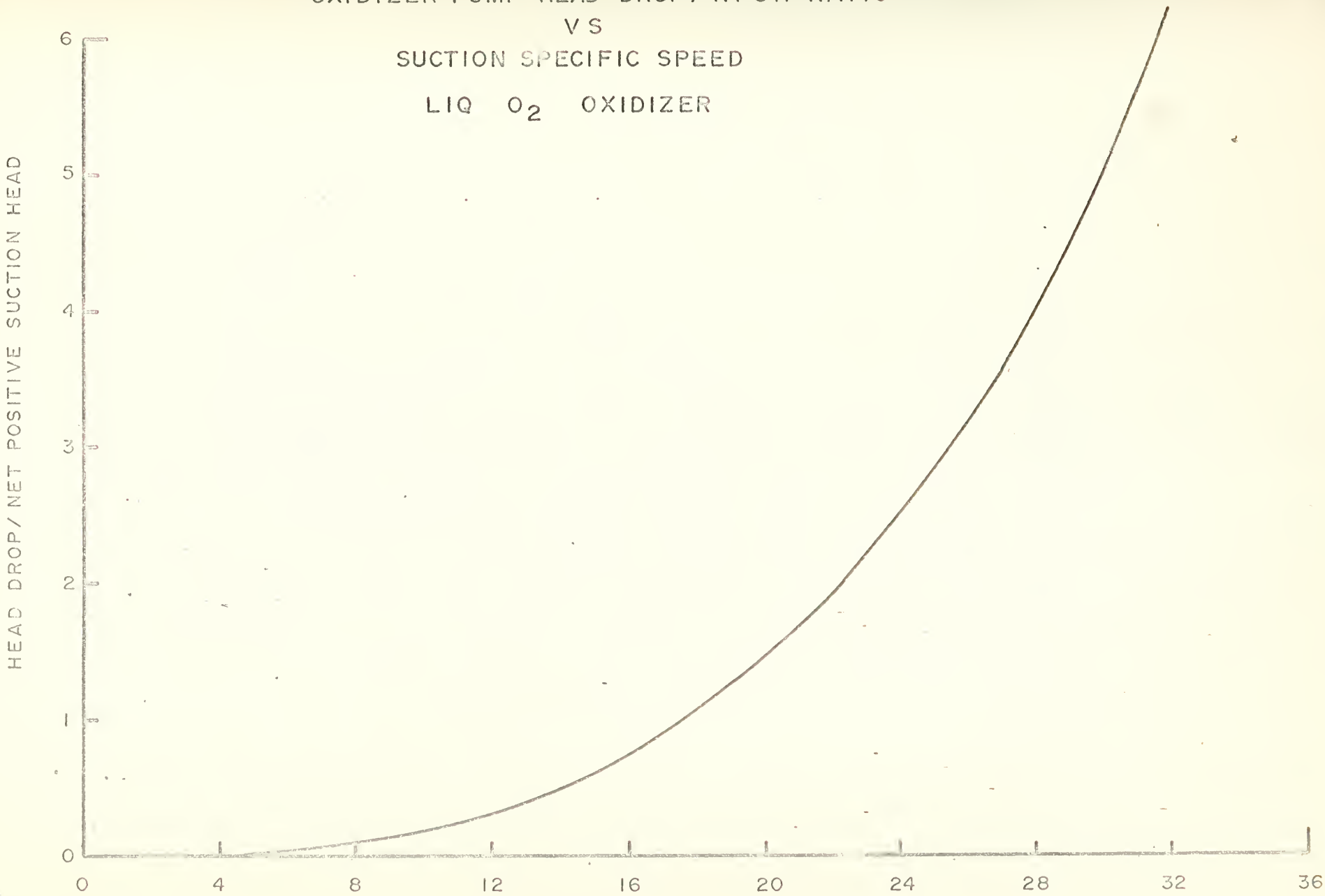


FIGURE C-2

OXIDIZER PUMP HEAD DROP / NPSH RATIO
VS
SUCTION SPECIFIC SPEED
LIQ O_2 OXIDIZER



APPENDIX D

PHYSICAL PROPERTIES OF HYDROGEN

and

ASSOCIATED PROPELLANTS

TABLE D-I

SATURATION CONDITION

TEMP.	PRESS.	DENSITY LB/FT^3		ENTHALPY BTU/LB			ENTHOPY BTU/LB/°F		
°F	psia	LIQUID	VAPOR	LIQUID	VAPOR	EVAP.	LIQUID	VAPOR	EVAP.
-435.5	.960	4.82	.00812	-	-	-	-	-	-
-431.3	2.838	4.712	.02000	-	-	-	-	-	-
-427.6	6.601	4.591	.0411	-	-	-	-	-	-
-424.0	13.040	4.459	.0748	86.0	282.3	195.0	2.05	7.51	5.46
-420.6	22.910	4.307	.1249	94.8	284.8	189.9	2.28	7.07	4.79
-416.8	37.000	4.135	.1968	104.9	285.9	181.3	2.51	6.70	4.19
-412.9	56.200	3.935	.2980	116.9	287 1	170.3	2.76	6.40	3 64
-409.5	81.550	3.697	.4413	131.0	286.0	155.0	3.03	6.11	3.08
-405.9	114.300	3.390	.6540	148.1	280.8	132 5	3.33	5 78	2.45
-404.0	134.000	3.175	.7900	157.8	273.3	115.9	3.50	5.57	2.07
-402.8	156.100	2.914	1.018	168.3	268.2	91.3	-	-	-
-399.9	188.000	1.88	1.884	250.2	250.2	0	4.59	4.59	0

HYDROGEN (H_2) PROPERTIES

TABLE D-II

SATURATED CONDITION

TEMP.	PRESS.	DENSITY LB/FT^3		ENTHALPY BTU/LB			ENTHROPY BTU/LB./$^\circ$F		
$^\circ$F	psia	LIQUID	VAPOR	LIQUID	VAPOR	EVAP.	LIQUID	VAPOR	EVAP.
-297.42	14.7	71.20	.2766	8.89	100.6	91.69	.0594	.6247	.5653
-288.7	23.52	69.70	.4431	12.30	101.3	88.93	.0797	.6003	.5206
-279.7	36.75	68.10	.6599	16.00	102.2	86.18	.1009	.5797	.4788
-270.7	54.831	66.50	.9989	19.70	103.0	83.25	.1206	.5613	.4406
-261.7	79.086	64.70	1.3720	23.60	103.7	80.10	.1400	.5447	.4047
-252-7	110.397	62.80	1.912	27.50	104.3	76.78	.1588	.5297	.3709
-243-7	149.94	60.70	2.55	31.30	104.7	73.35	.1753	.5147	.3394
-234.7	198.597	58.80	3.38	35.20	104.9	69.64	.1920	.5006	.3090
-225.7	257.544	56.40	4.34	39.40	104.8	65.42	.2070	.4869	.2800
-216.7	326.781	54.00	5.51	43.70	104.3	60.64	.2250	.4744	.2500
-207.7	410.13	51.10	6.99	48.40	103.1	54.70	.2430	.4603	.2170
-198.7	505.68	47.90	9.05	53.70	100.3	46.60	.2630	.4416	.1790
-189.7	620.34	42.30	12.7	60.81	94.61	33.80	.2890	.4141	.1250
-182.00	730.59	-	26.8	78.36	78.36	0	.3516	.3516	0

OCYGEN (O_2) PROPERTIES

TABLE D-III

SATURATED CONDITION

TEMP. °F	PRESS. psia	DENSITY LB/FT^3		ENTHALPY BTU/LB			ENTROPY BTU/LB/°F		
		LIQUID	VAPOR	LIQUID	VAPOR	EVAP.	LIQUID	VAPOR	EVAP.
-306.6	14.700	94.22	.3517	-44.336	27.189	71.525	10.336	18.725	8.389
-298.0	24.711	91.10	.5693	-40.926	28.326	69.252	10.691	18.384	7.693
-289.0	40.322	88.61	.8948	-37.563	29.131	66.694	11.027	18.047	7.020
-280.0	62.651	85.80	1.3547	-34.152	29.794	63.946	11.364	17.758	6.394
-271-0	93.330	83.20	1.9725	-30.789	30.316	61.105	11.653	17.474	5.821
-262.0	134.093	80.50	2.8230	-27.426	30.505	57.931	11.927	17.195	5.268
-253-0	186.690	77.60	3.9324	-24.016	30.458	54.474	12.193	16.929	4.736
-244.0	252.840	74.88	5.3889	-20.652	29.984	50.636	12.420	16.640	4.220
-235.0	334.131	70.51	7.2946	-17.242	29.084	46.326	12.671	16.375	3.704
-226.0	432.327	66.14	9.7968	-13.831	27.331	41.162	12.903	16.067	3.164
-217.0	548.898	60.53	13.3224	-9.900	24.821	34.721	13.145	15.717	2.572
-208.0	684.726	51.79	19.1256	-5.968	19.468	25.436	13.396	15.215	1.819
-200.8	808.500	33.07	29.4278	-	0	-	14.244	14.244	0

FLUORINE (F_2) PROPERTIES

FIGURE D-1

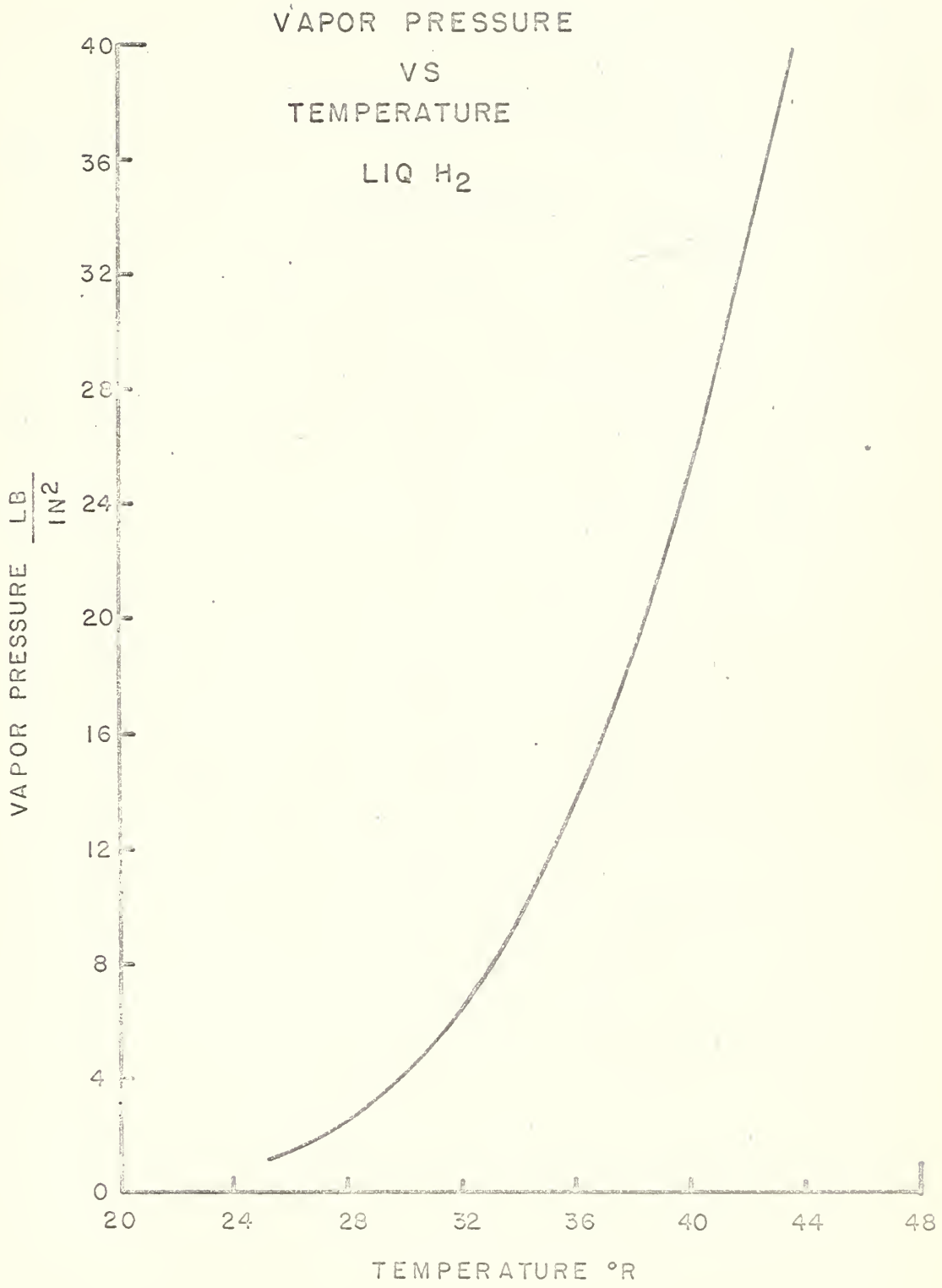

VAPOR PRESSURE
VS
TEMPERATURE
LIQ H_2
VAPOR PRESSURE $\frac{LB}{IN^2}$
0
4
8
12
16
20
24
28
32
36
40
20
24
28
32
36
40
44
48
TEMPERATURE °R

FIGURE D-2

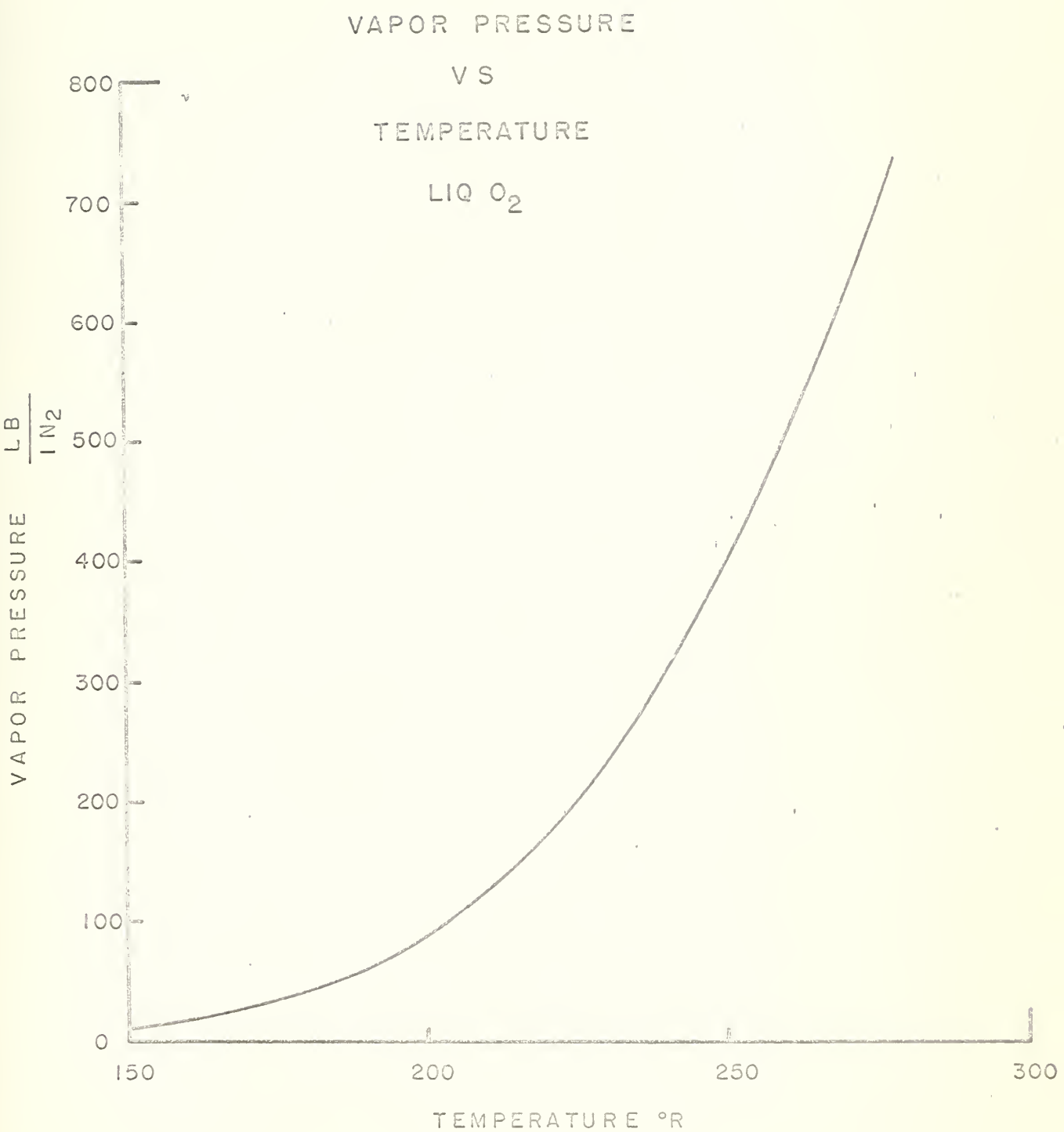

VAPOR PRESSURE
VS
TEMPERATURE
LIQ O_2
VAPOR PRESSURE $\frac{LB}{IN_2}$
800
700
600
500
400
300
200
100
0
150
200
250
300
TEMPERATURE °R

FIGURE D-3

VAPOR PRESSURE
VS
TEMPERATURE

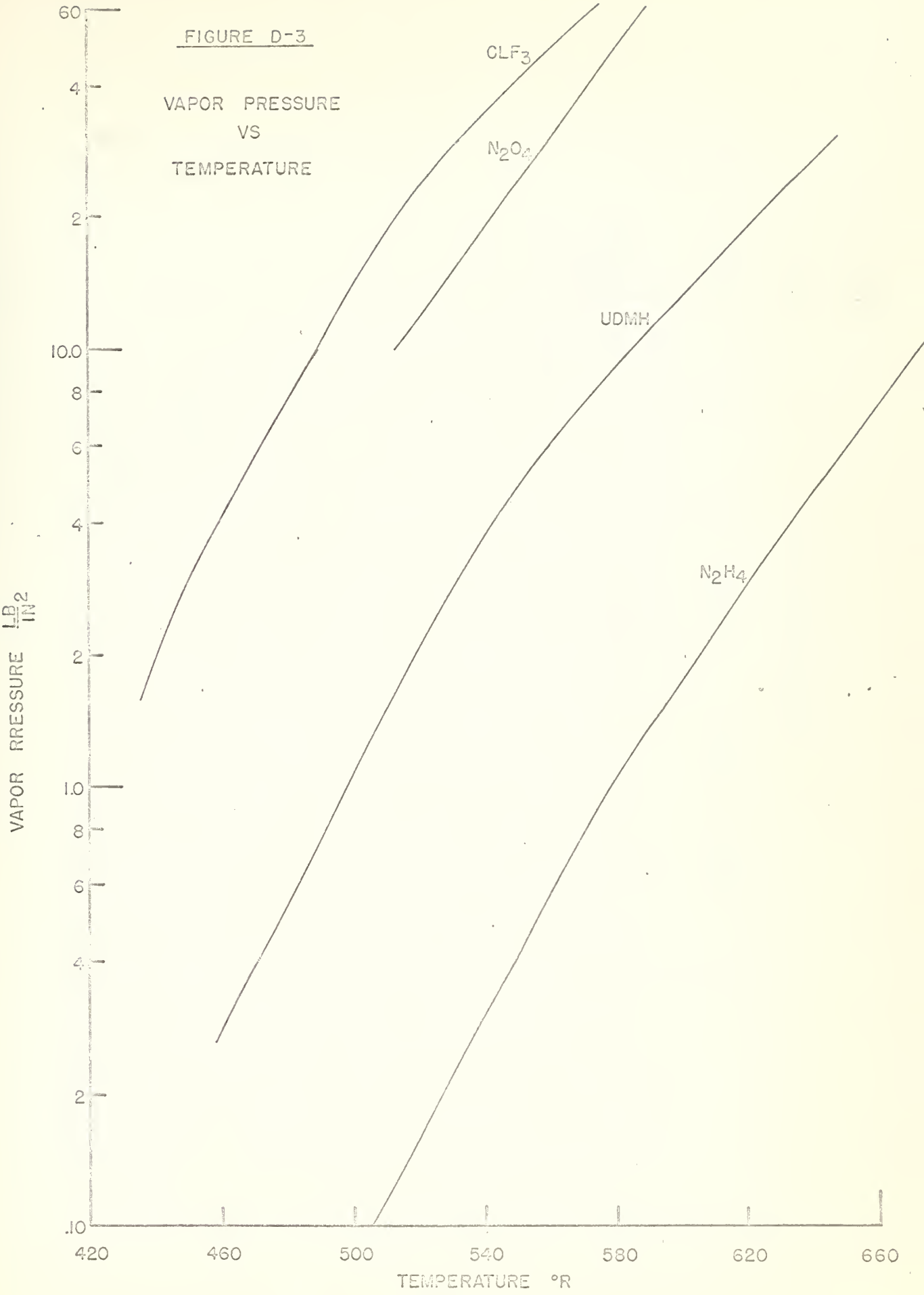
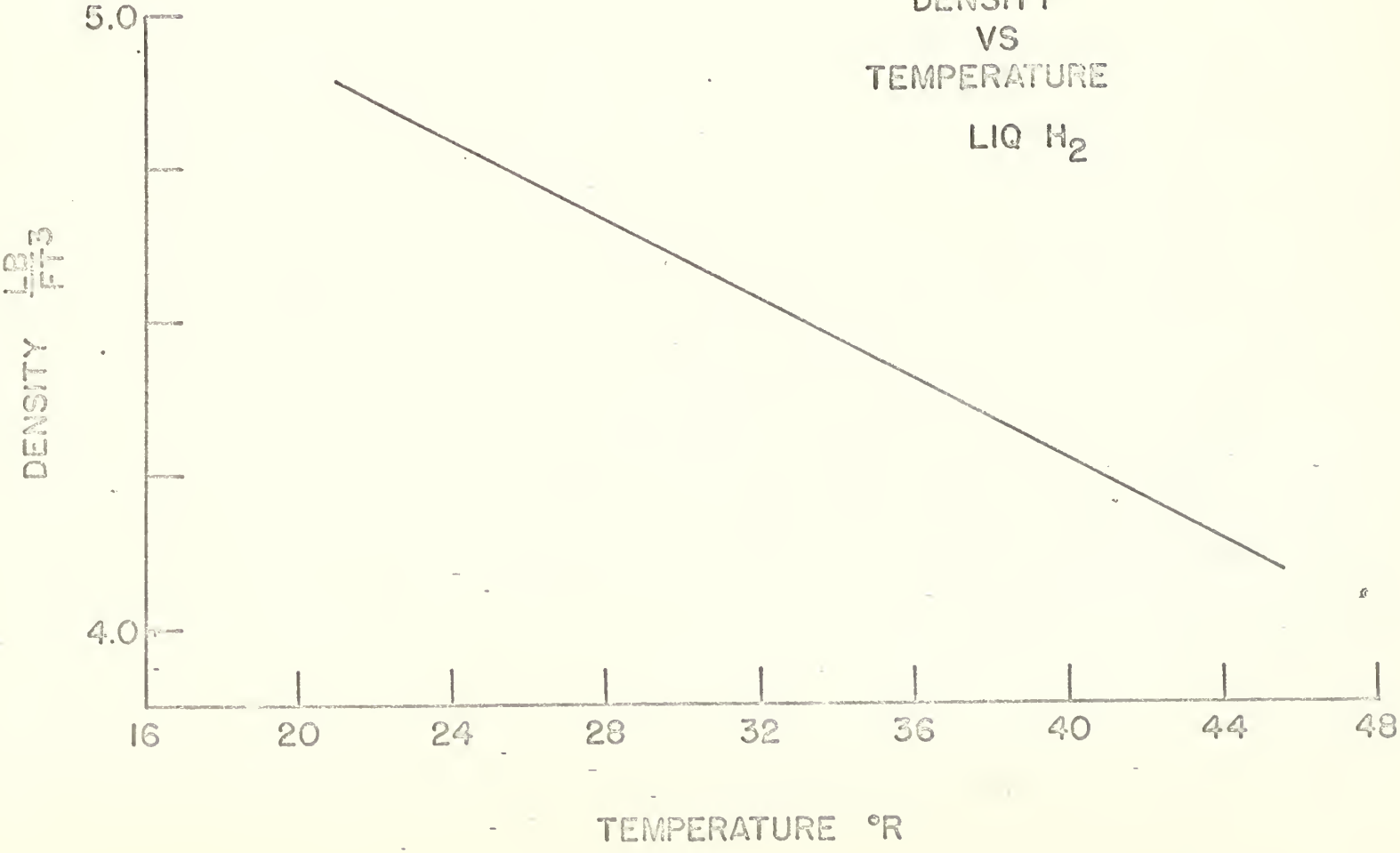


FIGURE D-4

DENSITY
VS
TEMPERATURE

LIQ H_2



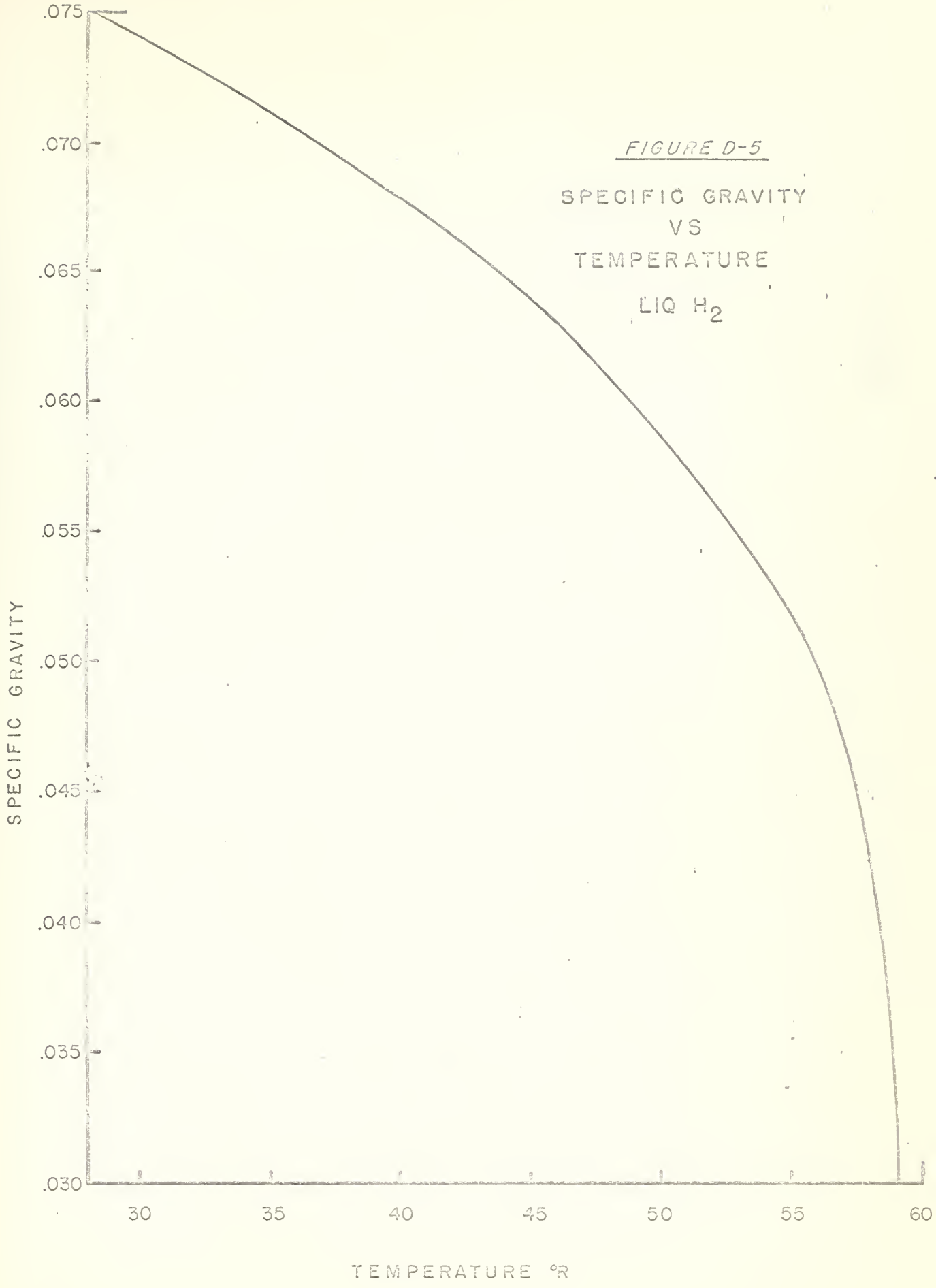


FIGURE D-5

SPECIFIC GRAVITY
VS
TEMPERATURE
LIQ H_2

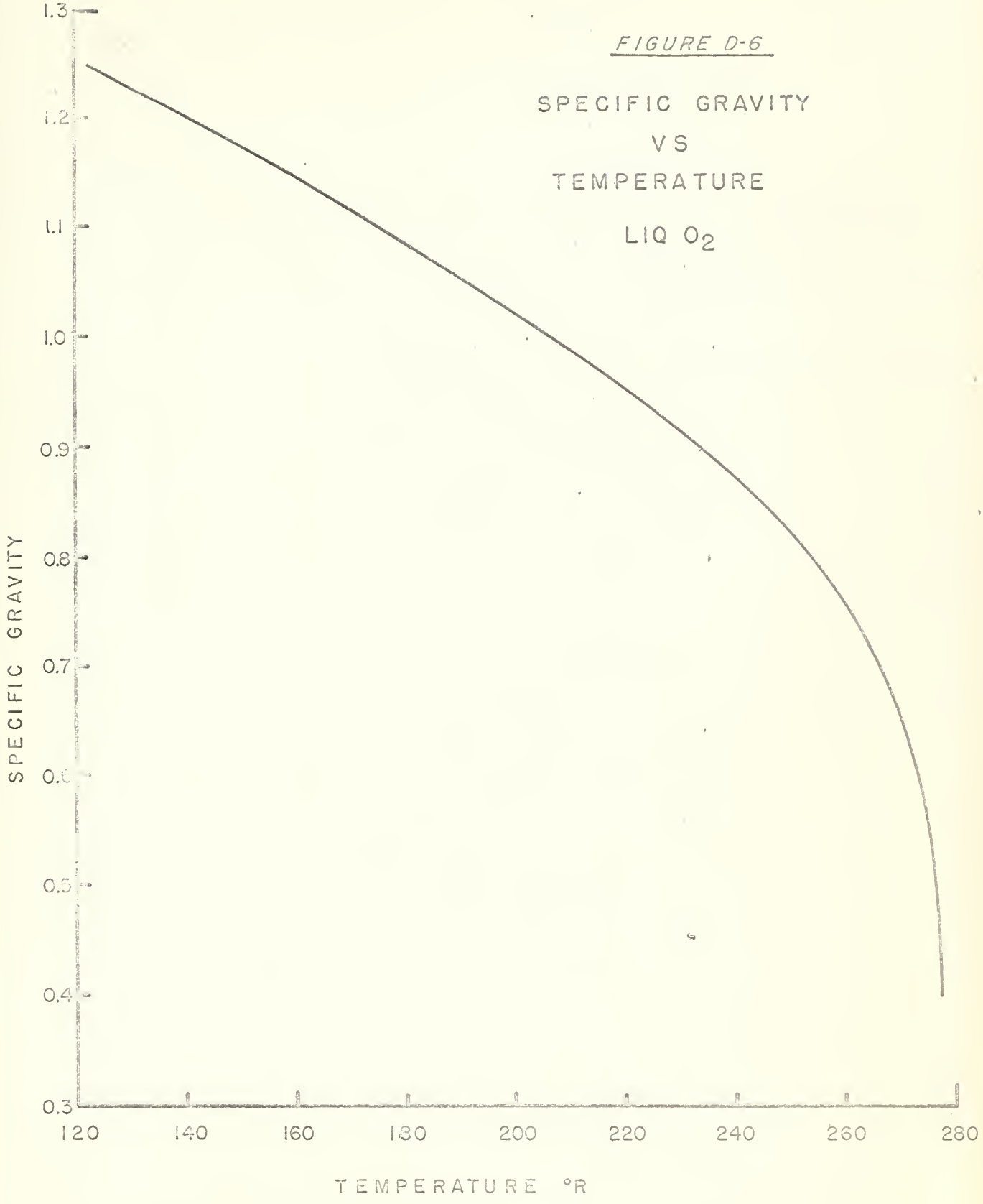


FIGURE D-6
SPECIFIC GRAVITY
VS
TEMPERATURE
LIQ O_2
SPECIFIC GRAVITY
1.3
1.2
1.1
1.0
0.9
0.8
0.7
0.6
0.5
0.4
0.3
120
140
160
180
200
220
240
260
280
TEMPERATURE °R

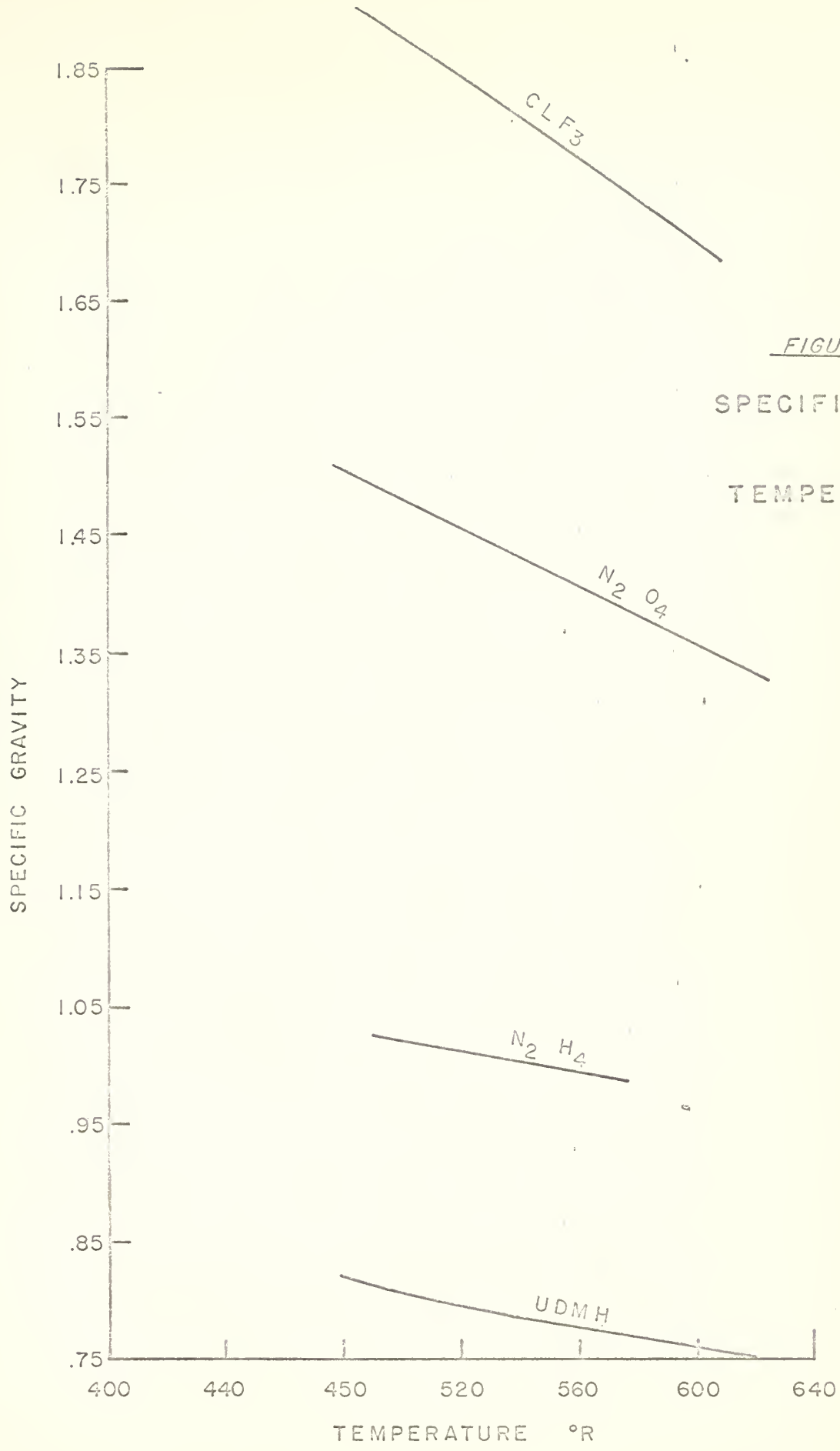
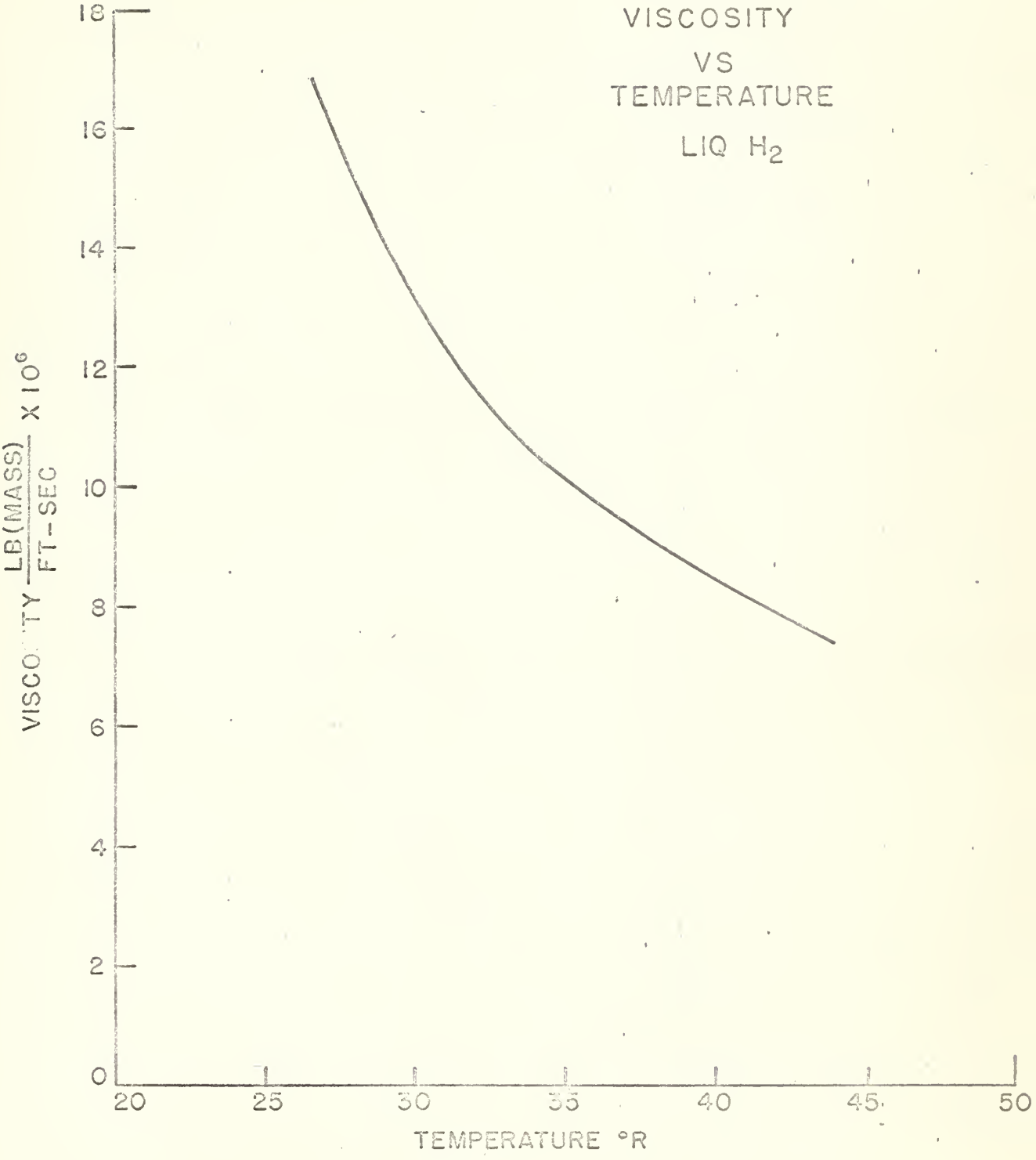


FIGURE D-7

SPECIFIC GRAVITY
VS
TEMPERATURE

FIGURE D-8

VISCOSITY
VS
TEMPERATURE
LIQ H_2



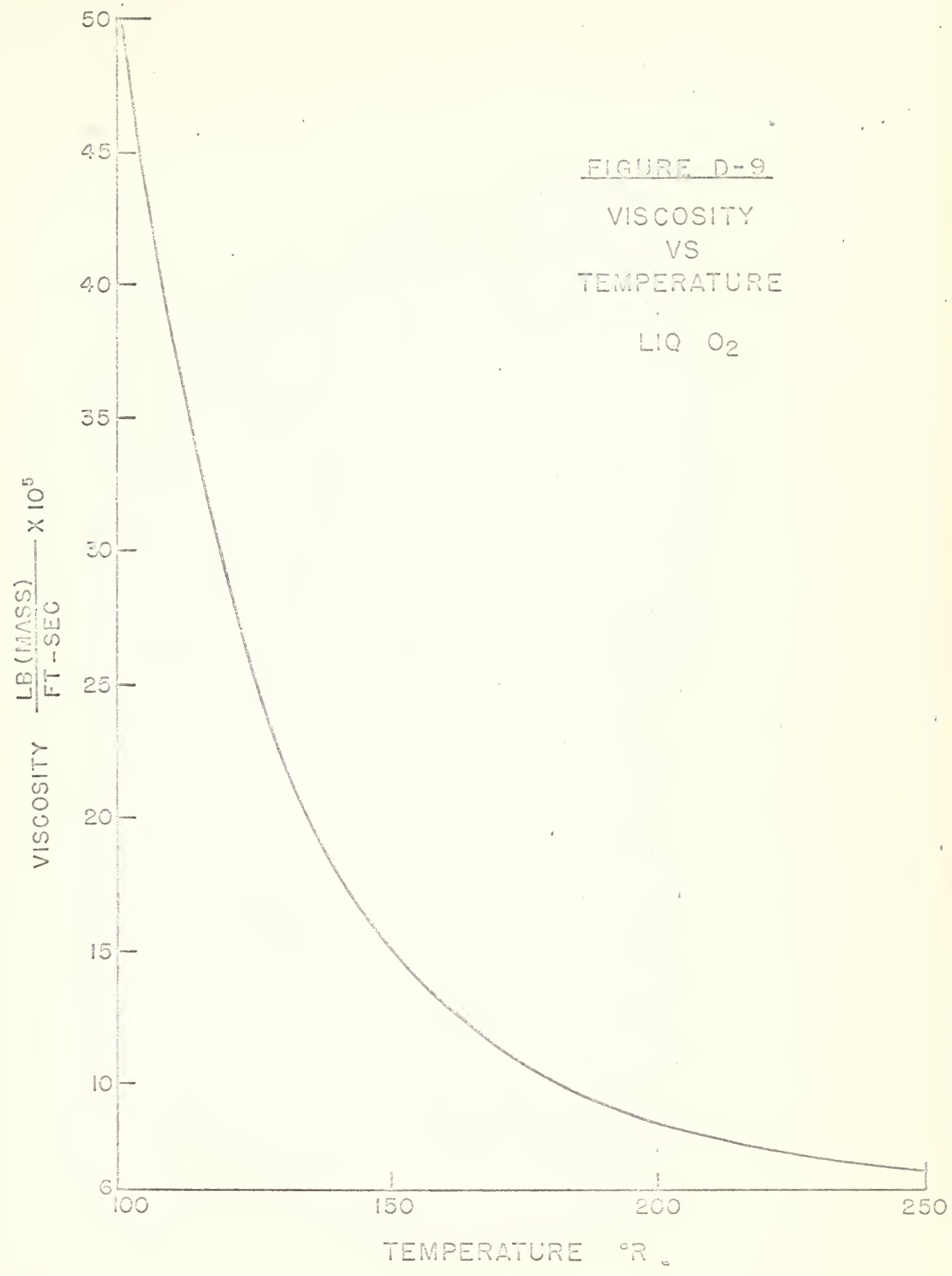


FIGURE D-9

VISCOSITY
VS
TEMPERATURE

LIQ O_2

50
45
40
35
30
25
20
15
10
5
VISCOSITY (poise) x 10³

FIGURE D-10

VISCOSITY
VS
TEMPERATURE

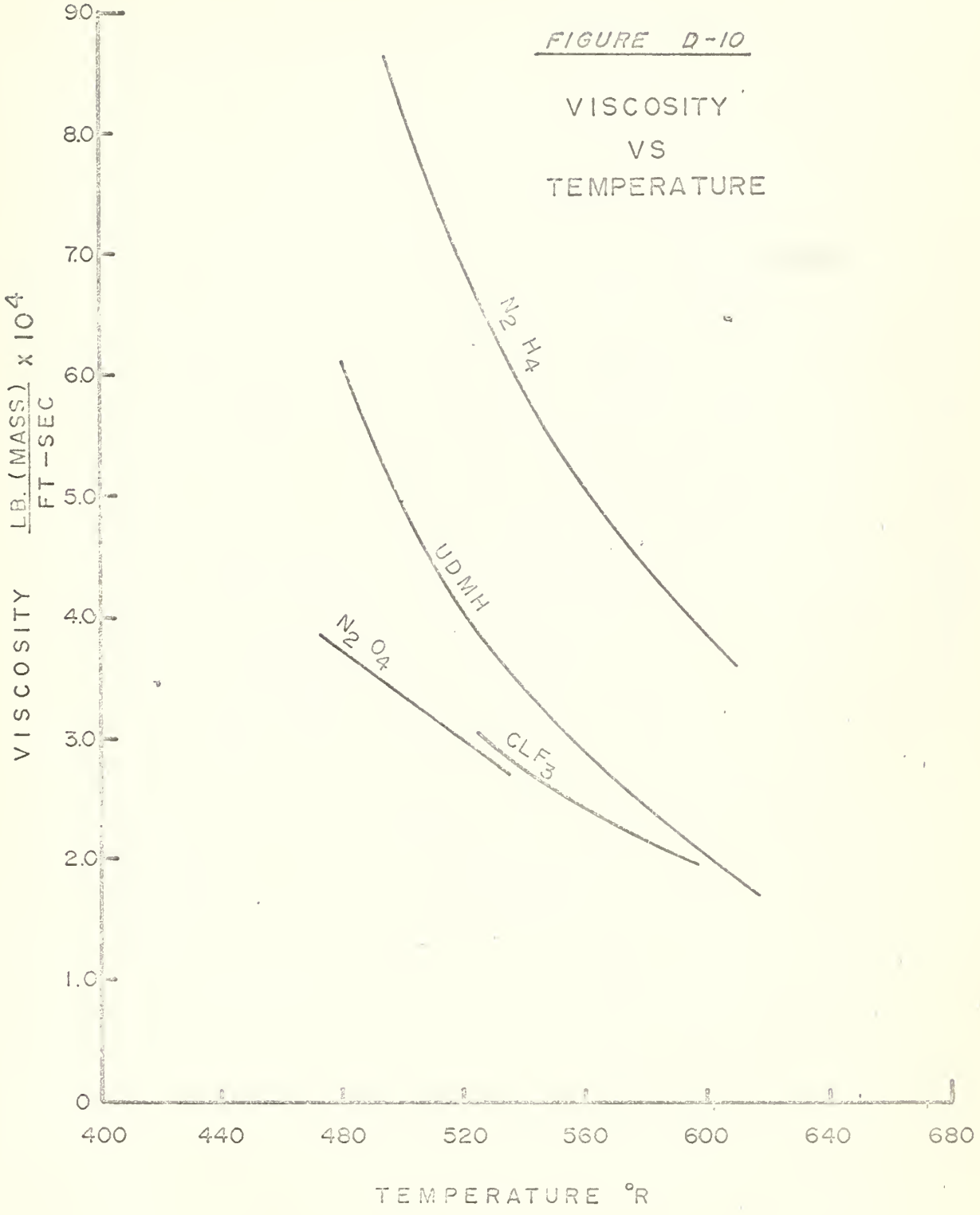


FIGURE D-11

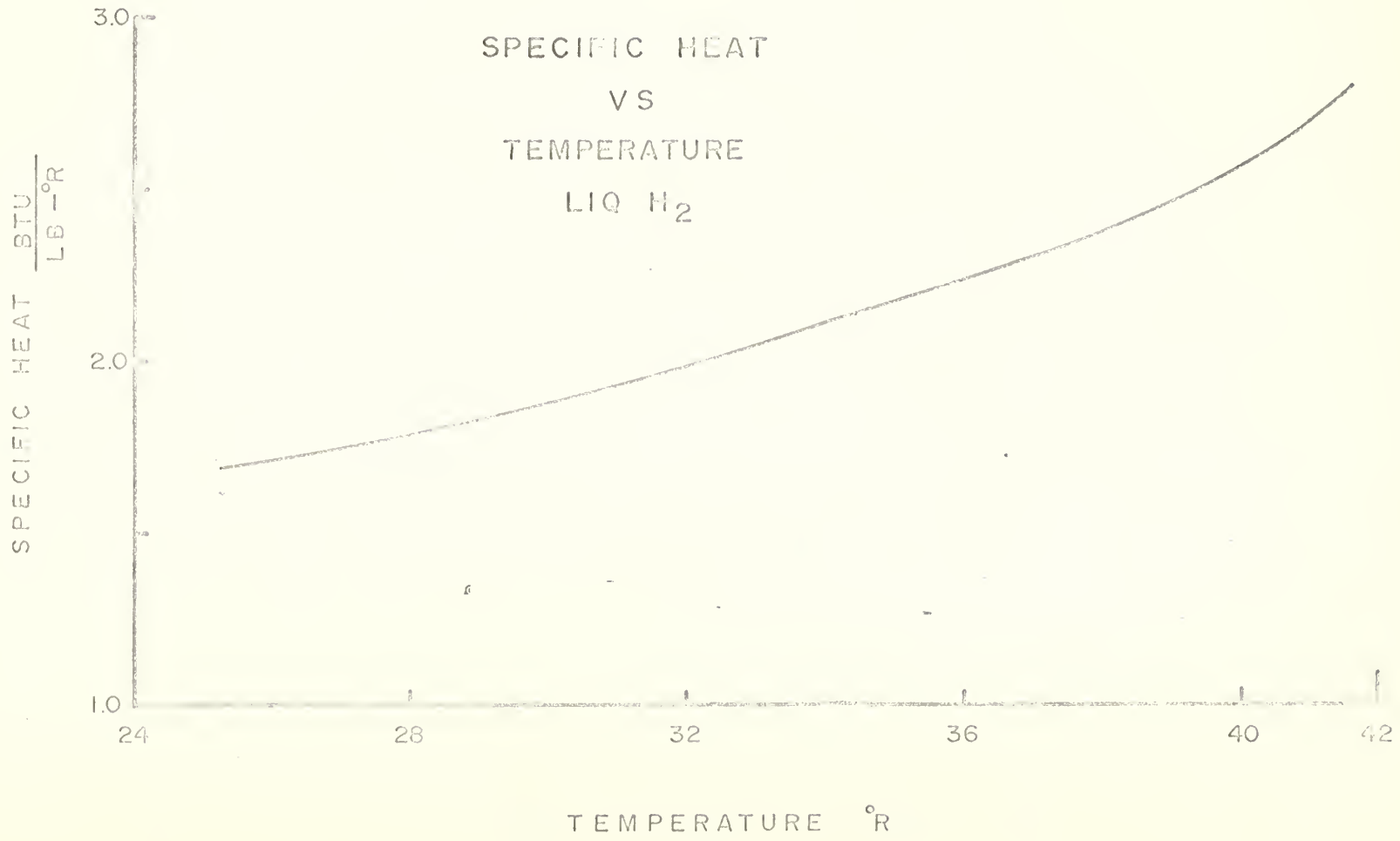
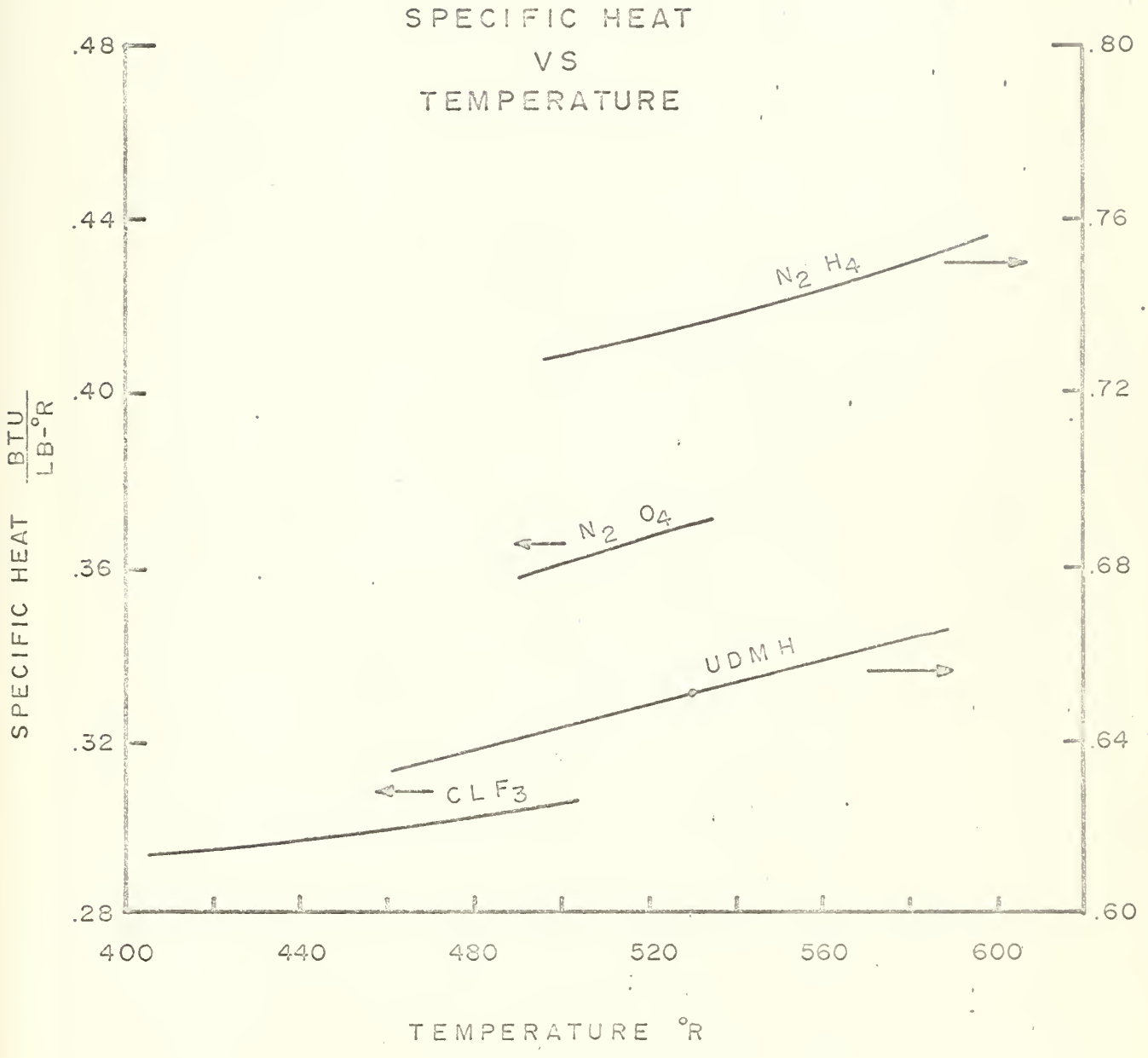

SPECIFIC HEAT
VS
TEMPERATURE
LIQ H_2
SPECIFIC HEAT $\frac{BTU}{LB-°R}$
3.0
2.0
1.0
24
28
32
36
40
42
TEMPERATURE °R

FIGURE D-12


SPECIFIC HEAT
VS
TEMPERATURE
SPECIFIC HEAT BTU / LB-°R
.48
.44
.40
.36
.32
.28
.80
.76
.72
.68
.64
.60
400
440
480
520
560
600
TEMPERATURE °R
N2 H4
N2 O4
UDMH
CLF3

thesC3384
Liquid hydrogen :
3 2768 002 09137 3
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